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# ENGINE SYSTEMS OWNERSHIP COST REDUCTION IN AIRCRAFT PROPULSION SUBSYSTEM INTEGRATION (APSI)

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AUGUST 1975

INTERIM REPORT FOR PERIOD APRIL 1973 - AUGUST 1975

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This report has been reviewed by the Information Office, (ASD/OIP) and is releasable to the National Technical Information Service (NTIS). At NTIS, it will be available to the General Public, including Foreign Nations.

This technical report has been reviewed and is approved for publication.

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This report summarizes Teledyne CAE's experience and conclusions in developing reduced-cost adaptive components, airframe interface requirements, and integrated systems plans under the AFAPL-sponsored Aircraft Propulsion Subsystem Integration (APSI) program. The APSI program is directed to evolving the Teledyne CAE Model 555 ATEGG into a turbofan Joint Technology Demonstrator Engine (JTDE).

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Reduced cost of engine ownership is one major objective of the APSI program. It was, therefore, deemed advisable to summarize that task into this document. This report is submitted to the USAF Aero Propulsion Laboratory (AFAPL) in partial fulfillment of Contract Data Requirements List Item A008, Final Report, of Contract F33657-73-C-0620.

Section 2.0 of this report provides an overview of engine ownership costs, as perceived and evaluated by Teledyne CAE, in accomplishing APSI-program tasks. These tasks drove the evolution of the design-to-life-cost (DTLC) methodology described herein. This section identifies the cost elements; describes approaches for their evaluation and reduction; and introduces a notational logic that equates life cycle cost elements.

Section 3.0 describes the application of the cost reducing methodology, including the results of iterating the baseline and scaled engines and applying Design-to-Cost (DTC) methods during their design. The benefits of conducting airframe company coordination are reviewed, and summarized costs for the baseline engine and scaled configurations of the baseline engine are included.

Section 4.0 advances 18 specific cost reduction topics, ranging from materials technology through candidate cost-reducing components to possible savings resulting from engine model specification changes. In effect, these topics can singly, or in combination, lead to further cost reducing iterations of the baseline engines.

Section 5.0 addresses the subject of engine ownership cost reduction as an evolving methodology. Comments are offered on its present status and objectives. Some suggestions for improving the precision and usefulness of Design-to-Life-Cost methods are submitted.

# FOREWORD

This report constitutes an advanced segment of the Final Technical Report of the Aircraft Propulsion Subsystem Integration Contract, performed by Teledyne CAE under the sponsorship of the United States Air Force/Aero Propulsion Laboratory, Wright Patterson Air Force Base, Dayton, Ohio. The work described herein was accomplished under Contract No. F33657-73-C-0620 with Mr. Robert Panella, AFAPL Project Engineer.

The work was performed by the technical staff of Teledyne CAE under the direction of Mr. Wesley Knight, Project Engineer. This report was authored and edited by Messrs. Alfred Gabrys and William Wagner of Teledyne CAE. It specifically addresses the system cost reduction aspect of aircraft engine design and technology.

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# Introduction

# SECTION 1.0 - INTRODUCTION

## 1.1 Purpose

This report summarizes Teledyne CAE's experience and conclusions in developing reduced-cost adaptive components, airframe interface requirements, and integrated systems plans under the AFAPL-sponsored Aircraft Propulsion Subsystem Integration (APSI) program.

Reduced cost of engine ownership is one major objective of the APSI program. It was, therefore, deemed advisable to summarize that task into this document. This report is submitted to the USAF Aero Propulsion Laboratory (AFAPL) in partial fulfillment of Contract Data Requirements List Item A008, Final Report, of Contract F33657-73-C-0620.

## 1.2 Discussion of Contents

Section 2.0 of this report provides an overview of engine ownership costs, as perceived and evaluated by Teledyne CAE, in accomplishing APSI-program tasks. These tasks drove the evolution of the design-to-life-cost (DTLC) methodology, described herein. This section identifies the cost elements; describes approaches for their evaluation and reduction; and introduces a notational logic that equates life cycle cost elements.

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# Section 2.0
# Propulsion Systems - Cost Of Ownership

# SECTION 2.0 - PROPULSION SYSTEMS - COST OF OWNERSHIP

## 2.1 Mission Analysis

The APSI program Statement of Work (SOW), clearly defined the objective of reducing the cost of engine ownership. During Phase I, (Design Definition) task emphasis included assessment of:

> "...the effects of the interrelationships between engine component cost and performance factors on overall aircraft system cost/engine component cost..."

The SOW also recognizes the need for a detailed mission scenario as a prerequisite for evaluating tradeoffs, and goes on to require that:

> "The contractor shall analyze USAF requirements...and select... two (2) missions.
>
> "...Missions selected shall be representative...and will match the Teledyne CAE ATEGG scalable thrust class (500 to 10,000 pound). ...Optimization of the engine design shall be based on minimum overall system cost..." (1)

In pursuing the requirement, to consider aircraft/engine cost relationships, it quickly became evident that a structured program of coordination with airframe contractors was necessary. Accordingly, Teledyne CAE conducted discussions with a number of airframers and then subcontracted studies relating to different missions to two airframers. This support provided two valuable services, i.e.:

1. It permitted accomplishment of an on-going dialogue between Teledyne CAE's engine design and the airframer's engine/aircraft integration specialists.

2. It fostered understanding of the total scope and diversity of engine ownership costs by directing attention to the impact that engine design has on the airframe.

As the APSI program progressed, a systematic approach to engine ownership cost evolved at Teledyne CAE. It commenced with consideration of the acquisition cost of production engines based using the the Design-to-Cost (DTC) approach. Subsequently the program matured into a definition of a set of birth-to-grave elements of engine ownership cost with a consistent emphasis on the need to address cost on the drawing board.

## 2.2 Elements of Systems Cost

As APSI systems cost reduction studies continued, a lexicon developed which identified engine design influenced costs of engine ownership by major and minor elements. It also became necessary to identify the time in an engine's life cycle at which costs of the various elements would commence, continue, and phase out. In other words, a matured cost analyses method will evaluate the engine owners' time-phased "cash flow".

Identification by major elements is necessary because more than one DOD activity is involved in funding an engine life cycle. Time-phasing is important because of the combined, but partly off-setting, influences of present value and escalation value.

The elements that contribute to or "drive" engine life cycle costs are numerous but each can be expressed by an equation. For this reason, a logical notation scheme deserved to be developed. Some cost evaluations are amenable to nominal accomplishment; however, the more complex problems, and most importantly the need to iterate and test sensitivity, will require a computer capability. Based on these criteria, a hierarchial notation scheme, was assembled to serve those purposes.

The notation scheme is based on the following logic, and the preliminary structure is shown in Figure 2.2-1:

| | | |
|---|---|---|
| CLO [X] I | = | Life Cycle Cost (LCC) of ownership for the model X engine and $i^{th}$ iteration of its LCC evaluation. It is expressed (unless otherwise noted) in $ at a "datum period" (see below) value, and represents the total area under the curve where time is the independent and cost is the dependent variable (i.e., CLO = sum of CJO's). |
| CJO | = | Cost to the engine owner in the "J"th period of the engine life cycle; where J = 1 (the datum period), 2, 3, 4,..., H (the horizon period). Unless otherwise noted, present value and escalation (+ or −) adjustments commence at J = 2. |
| J | = | The "J"th period. For time functions, the engine life cycle is divided into periods which could be years (an upper limit) or months (a lower limit). For most purposes, the fiscal quarter should be adequate. |
| "Datum" | = | The first period in the life cycle. Escalation and Present-Value coefficients have the value 1.0 in this period. Other cost-input estimates can be adjusted to this value. |

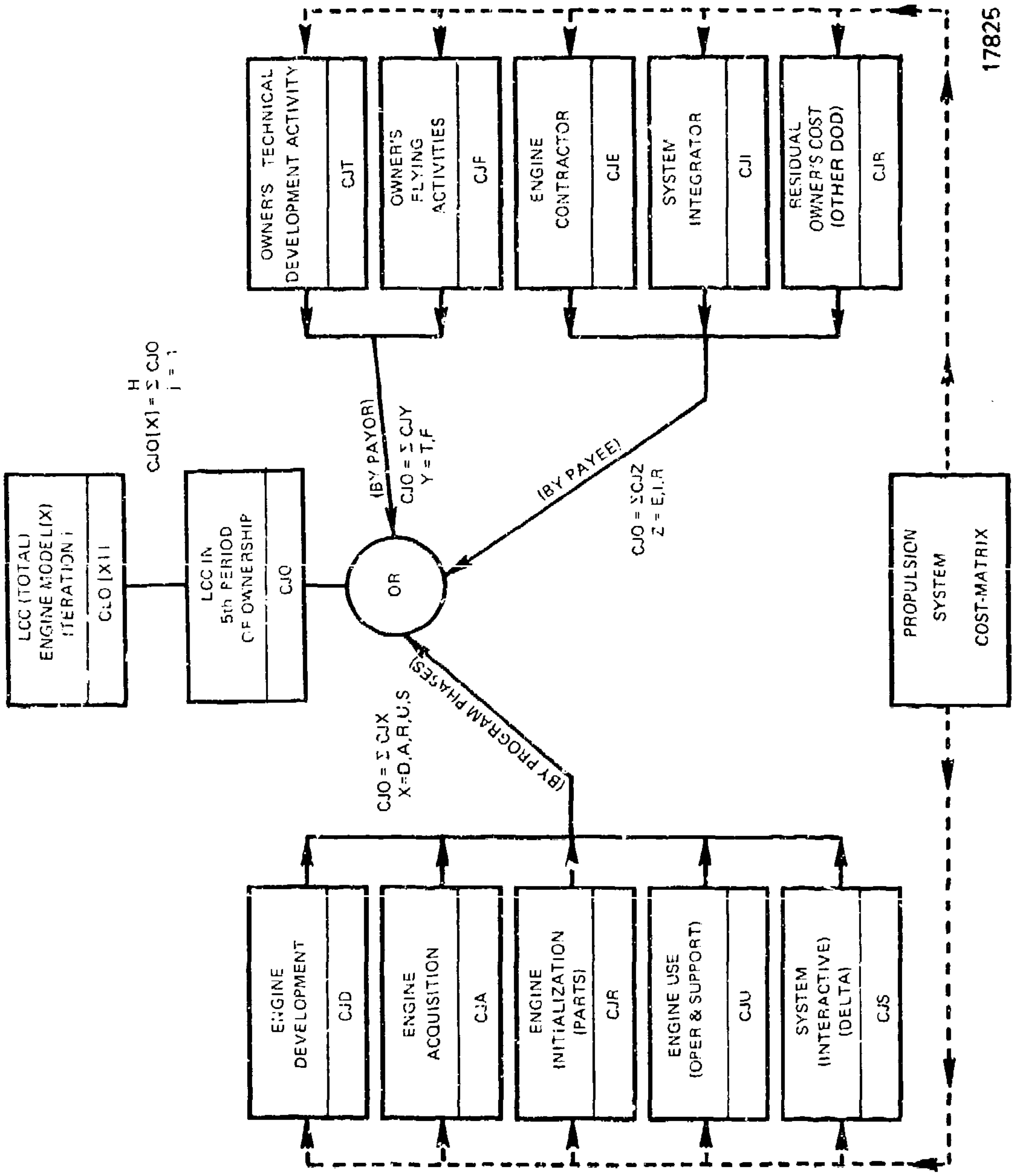



Figure 2.2-1. Engine Model Iteration.

| | | |
|---|---|---|
| "Horizon" | = | The last period in the life cycle. Twenty-eight yearly (or 100 quarterly) periods are typical for a trainer engine, and 15 yearly periods may be representative of an RPV engine. |
| CLD, CLA | = | Life Cycle Cost (total) of ownership for the development and acquisition phases respectively. |
| $P\left[CLD \leq X\right]$ | = | A probability statement utilized when technical-risk (See Sections 2.2-2 and 5.2), or when other confidence-bounded estimates are addressed. |

The importance of "time-of-cost-incurrence" deserves reiteration; first because of the purely economic aspect, and secondly because of the complex nature of technology and funding. To illustrate this point, Figures 2.2-2 and 2.2-3 develop lower elements of the LCC structure and suggest their typical duration within the engine life cycle.

The following subsections identify and discuss those elements of Engine Systems Cost that were recognized and addressed in the APSI program. While the first category, Specification-Imposed-Cost, is not in itself time-phased; it can have an enduring influence across the phases of the engine's life cycle.

## 2.2.1 Engine Specification Imposed Costs

The engine model specification is a cost driver because it sets the requirements for design, development and qualification of an engine model, and also stipulates the acceptance test requirements for production engines. Therefore, the APSI program SOW included a requirement to critique the cost impact of specifications (in Section 1.1.3, Reduced Cost Analysis) as:

> "...The contractor shall review the Applicable Military Specifications with respect to mission requirements. The impact on engine design cost of inappropriate requirements shall be assessed and specification changes shall be recommended to reduce cost..."

Three major sources of specification requirements were considered:

1. USAF - Developed Requirements:

   These are listed in MIL-E-5007D, Military Specification; Engines, Aircraft, Turbojet and Turbofan - General Specification for, 15 October 1973 (latest issue) where they are intended to serve as a general checklist and guideline - and not as set of exhaustive requirements to be applied to every engine for every application.

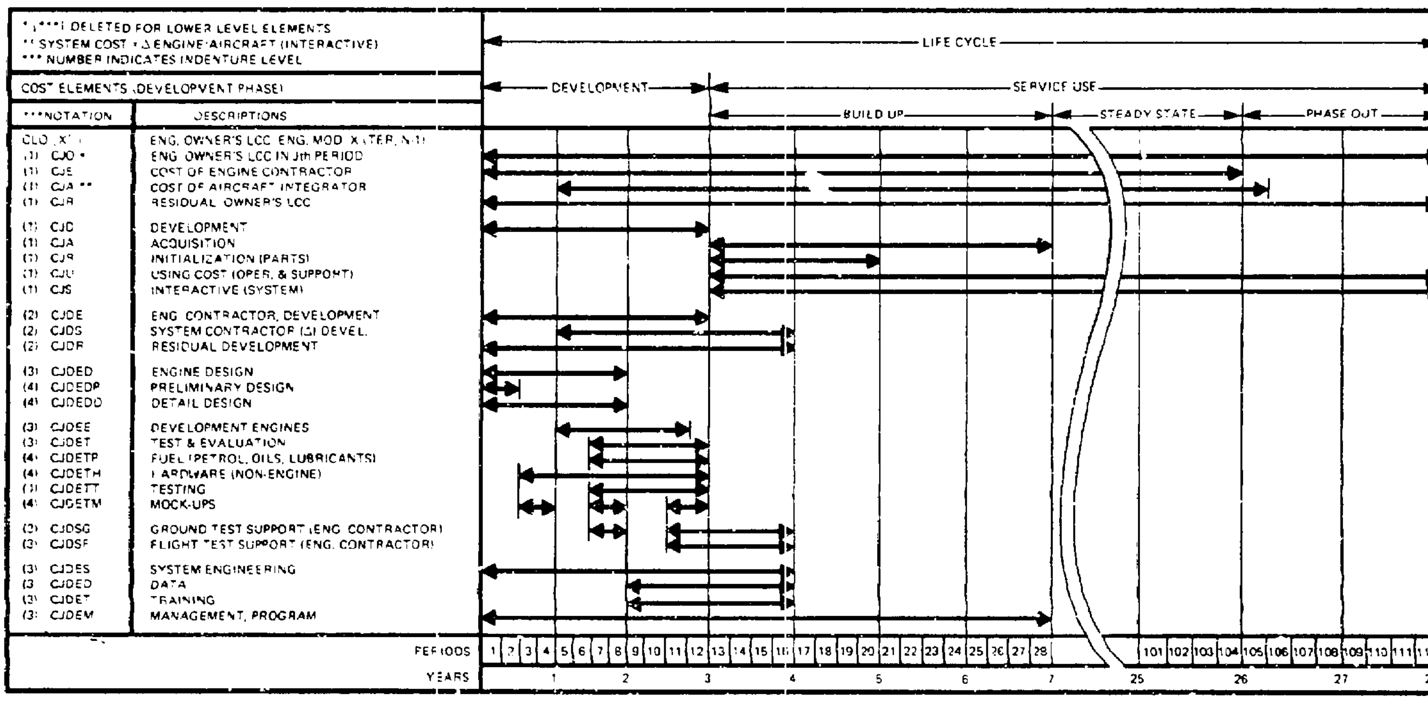



Figure 2.2-2. Cost Element Indenture/Duration: Typical Engine Development Phase.

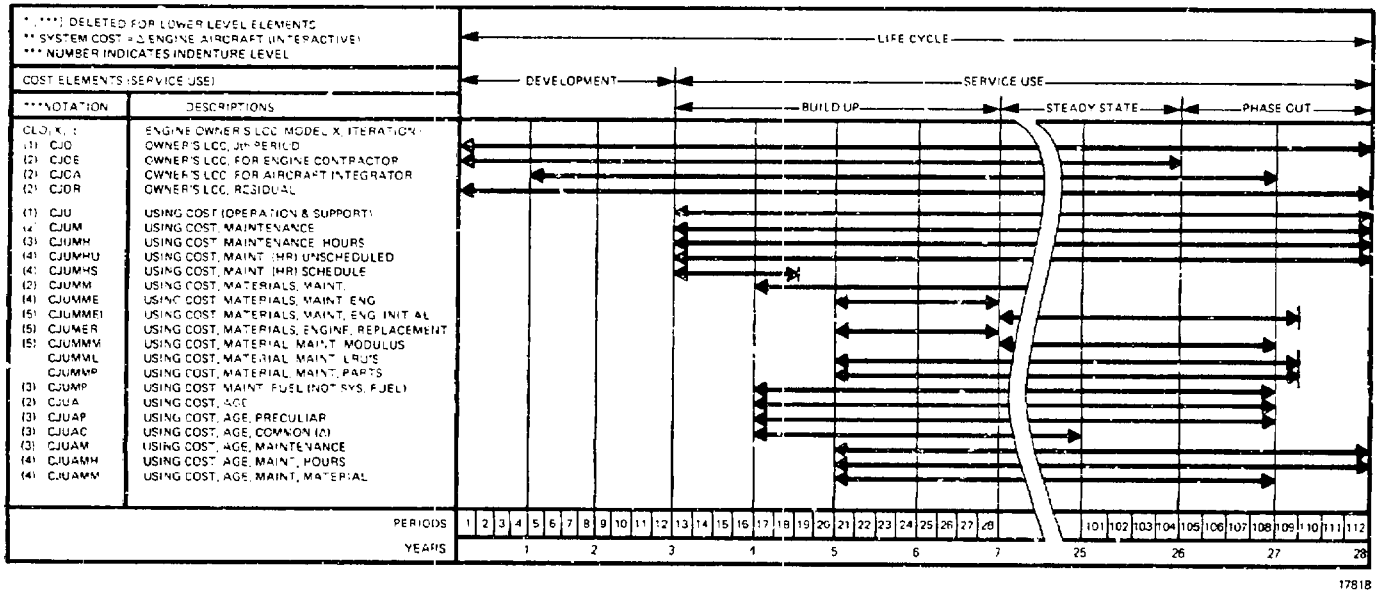



Figure 2.2-3. Cost Element Indenture/Duration: Typical Service Phase.

2. USN - Developed Requirements:

These are listed in AS 2684 - Military Specification; Engines, Aircraft, Turbojet and Turbofan, General Specification for, 30 December 1970 (latest issue). This document serves a purpose similar to MIL-E-5007D for Naval aircraft engine procurement.

3. Airframer - Developed Requirements:

These are usually imposed on Contractor Furnished Equipment (Engines) or CFE as opposed to Government Furnished Equipment (Engines) or GFE. CFE specifications usually contain (and derive from) military requirements. However, they also reflect, at times, requirements developed by the airframe contractor's specialist groups.

In undertaking a cost effectiveness critique of the foregoing specifications, three salient facts were considered, i.e.:

1. Teledyne CAE recognized that these specifications represent the accumulated experience of the military turbine-engine procurement and using activities. As such, the majority of their specific requirements have come into being in response to service-revealed problems, some applicable to specific engine types and some to engines in general.

2. The DOD specifications are specifically intended to serve as "checklists" and guidelines for negotiating a specific model specification. So, the cost impact is actually determined by the resulting model specification and the judgment exercised by the procuring activity and engine contractor.

3. Very few model specifications reflect blanket-use of DOD specifications; as evidenced by a review of current Teledyne CAE specifications for the J69/J100/J402 series of man-rated and non-man-rated engines.

As a result of the review process, Teledyne CAE also developed guidelines for the task of developing model specifications for APSI-derivative engines. These are addressed to the major sections of a general military (or contractor-equivalent) specification as follows:

a. Performance Considerations. Model Specifications should be reviewed for the cost impact of the operating envelope, particularly its "corners". Demonstrating a "corner" may become a "cause celebre" to the point that it can double the cost of a planned test series, without contributing to the owner's operational benefits. In most applications, the review will require a coordinated effort that involves the air vehicle designer, the engine designer and the using activity. The costs (of development and acquisition) incurred by advanced performance parameters (e.g., SFC) should be traded off against the ultimate benefits to the user.

b. Mechanical Design. The mechanical integrity requirements have a direct calcuable cost impact, and should be critiqued for the application. General requirements for "Structural Integrity" include Engine Pressure tests, Low Cycle Fatigue, Containment, etc. These translate into choices such as; material quality and type, forgings versus castings, rig test time and other considerations. Because they generally stem from prior, and often unfortunate experiences, they should be challenged with prudence. Nevertheless, these requirements should be modified as appropriate to the mission and application.

c. Reliability, Durability, Maintainability and Design Life. These parameters are related and often confused, yet they are generally the driving force in the maintenance and support cost elements of an engine's life cycle cost. For that reason alone, they each warrant first a clear, unequivocal, definition and then a careful evaluation based on design, material properties and intended use. We suggest the following definitions:

Design Life - The time, in hours or cycles, that an engine will operate and complete a stated number of missions between major overhauls.

Reliability - The probability that the engine will complete the design life period without a shut-down, or major failure.

Maintainability - (Better described as "Maintenance Index") The number of man-hours of maintenance per engine operating hour during the design life.

Durability - The probability that an engine, or part, undergoing periodic inspection at stated intervals will be capable of return to service. (This is more readily calculated for specific parts.)

d. Testing. The testing requirements of a model specification are usually referred to as Quality Assurance Provisions. They generally include development and qualification tests. They may include provisions for reliability and maintainability demonstration, and could address "Durability". Finally, they prescribe quality assurance in terms of the engine production acceptance test. This entire package deserves scrutiny because: when overspecified it raises the development cost, and, when underspecified, it can adversely affect the operation and support elements of life cycle cost.

e. Systems Engineering and Data. Teledyne CAE has recently seen RFP requirements containing Systems Engineering and Data requirements that promoted confusion and could have led to unnecessary cost in development programs. This usually occurs when the procurement is CFE and the contractor passes on all government specifications, without screening, to all of its sub-contractors. In any engine development program, the various work package and data requirements should be coordinated between the procuring activity, the air vehicle designer, and the engine designer.

f. Cost Trade-offs. In the process of critiquing specification requirements, the total cost/benefits of a specific design feature, test plan, or performance requirements should be determinable in quantitative measure. This task assumes the existence of a working "design-to-life-cycle" cost capability.

In summary, a systematic approach to specification cost impacts has been developed, with specific emphasis on MIL-E-5007D and AS 2684. Appropriate Teledyne CAE Specifications have been selected for the purpose of comparison, and subjects for detailed examination were identified. A typical set of specification-imposed cost subjects was subjected to the systems cost approach, which evolved during the program; with the results described in Section 4.2.18 of this report.

### 2.2.2 Development Costs

A systems cost methodology requires precise definitions, if its calculations and estimates are to be consistently understood and accepted. In the broadest sense, development costs consist of all expenditures necessary to bring a design to a state of producibility. In a practical evaluation, however, the conventions that exist in engine procurement should be considered.

In a feasibility evaluation, an engine development is normally funded after some investment such as; company IR & D/B & P, DOD exploratory research, etc. A specific development model is then undertaken with so-called 6.4/6.5 funding, and carried through to a model qualification test (MQT), often with demonstration milestones such as the preliminary flight rating test (PFRT). During APSI, we, therefore, developed the following definition for Development Costs:

CLD = Cost of Development (total) = All funds paid to develop, test and then demonstrate that a specific engine model is ready for production.

We also found that development costs have historically represented a small percentage of the explicit cost of engine ownership; however, engine development cost also has a profound impact on aircraft system cost. For example, the APSI studies identified the prospective savings in life cycle costs of one representative aircraft summarized in Figure 2.2.2-1. These resulted because the airframe was in a conceptual design phase and capable of being scaled down to benefit from improved engine performance while keeping the "mission" constant. The savings accrued in

| Engine | | Airframe | | |
|---|---|---|---|---|
| Parameter | Improvement | Cost Element | | Savings (% - 1974) |
| SFC | 0.01 (one "point") | Development | - | 14.05 |
| | | Acquisition | - | 85.95 |
| | | Maintenance | - | - |
| | | Percent of Savings | - | 100 |
| Weight | 0.01 (one percent) | Development | - | 12.27 |
| | | Acquisition | - | 19.62 |
| | | Maintenance | - | 68.11 |
| | | Percent of Savings | - | 100 |

Figure 2.2.2-1. Engine Influence on Airframe Cost.

airframe development, acquisition and in aircraft life-time fuel consumption. This relationship of engine to aircraft interactive cost is further discussed in Section 2.2.5.

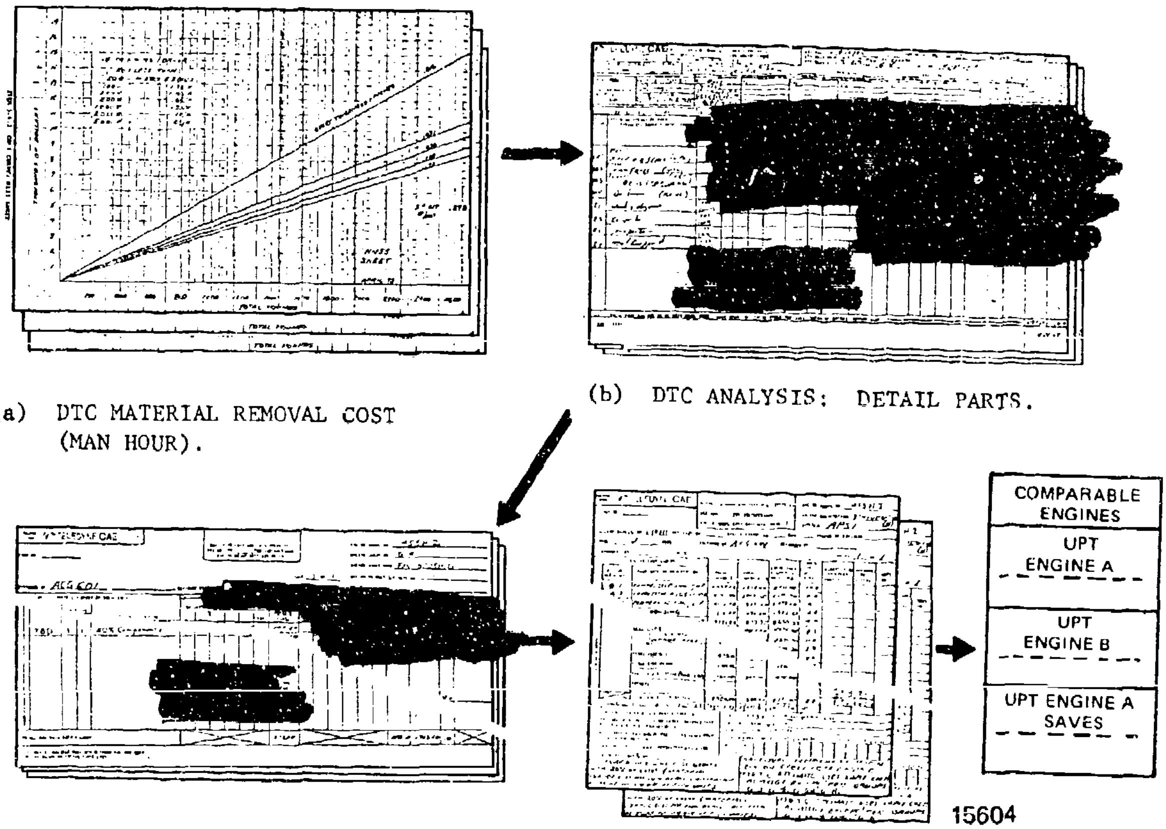
### 2.2.3 Engine Acquisition Cost and Design-to-Cost (DTC)

Summary

During the APL APSI program, Teledyne CAE developed an engineering approach that addresses the problem of engine costs and Design-to-Cost (DTC). The method was tried and tempered by real time, "on-the-drawing-board" use in the APSI component definition program. The company initiated DTC on APSI as a way of focusing engineering talent on the cost of small gas turbine engines. Teledyne CAE has since used DTC to good advantage in: the APL Low Cost Jet Fuel Starter design study; a replacement engine for the J69 series; and the current cruise missile engine development program.

The approach to DTC recognizes that acquisition and initialization are major contributing elements to the total cost of ownership. Also, initialization (stock set up) is a function of the number of parts in the engine and the cost of each part. Therefore, reducing acquisition cost and reducing engine complexity are mutually advantageous to total ownership cost.

The DTC method evolved for the JTDE engine and component use is briefly illustrated in Figure 2.2.3-1. A typical material removal curve that aids in selecting the proper material and manufacturing process for a detailed component is shown in Figure 2.2.3-1(a).




(a) DTC MATERIAL REMOVAL COST (MAN HOUR).

(b) DTC ANALYSIS: DETAIL PARTS.

(c) DTC: ENGINE COST GROUP.

(d) DTC: ENGINE MODELS.

Figure 2.2.3-1. DESIGN-TO-COST APPROACH.

Figure 2.2.3-1(b) illustrates an inseparable part DTC worksheet - completed and iterated while the part is on the design board. The accumulated costs for one engine "cost group" are listed in the DTC worksheet of Figure 2.2.3-1(c). These DTC "Cost Group" worksheets are then assembled and used to produce the engine model work sheet (Figure 2.2.3-1(d)) that aids the Design Group Leader during progressively more rigorous technical and management reviews.

The results of the DTC approach to engineered reductions of acquisition cost are compiled in the illustration for comparison of an APSI derivative to a current engine, where the APSI derivative saves 35 percent in engine costs.

[redacted] the APSI Program have been developed through [redacted] process. Original costing methods only provided data [redacted] for comparing percentages or trends of different or [redacted] This process was found to be inadequate when the cost reduction topics were investigated in detail. This eventually led to the establishment of a "Design-to-Cost" costing procedure to provide the necessary detail for accurate cost comparisons. The considerations addressed in the development of this method were:

- It must provide sufficient detail to be useful for the appropriate phase of design.

- The method should be compatible with manual as well as data processing compilation procedures.

- The approach must provide the [redacted] activity with sufficient information to [redacted] intent and contribute to achieving the cost [redacted]

The method selected is based on identifying the finite [redacted] engine cost at the part design level. During the design process, the cost of manufacturing (or acquiring) each part is compiled by finite elements. The elements include the man-hours for individual manufacturing operations, the material costs and the support costs (in man-hours and material) of inspection, tool support and certain overhead operations. Man-hour and material costs are entered as "burdened" values. Performance indices and anticipated scrap rates are included for each operation, or summarized for each part.

This method facilitates evaluating the cost consequence of changes in design and/or manufacturing operations. It also allows for estimating or specifying cost objectives as a function of production quantities and delivery rates. Burdened rates are used for material and labor but the burden (overhead and material handling costs) may vary as a function of the production quantity being estimated. However, all values are still in terms of cost and not price (i.e., they do not include G & A or Fee). A more detailed description of the costing procedure was presented in APSI briefing sessions.

In addition to the costing procedures described, data has been tabulated in the DTC Cost Compilation Manual to provide the means of deriving the material and fabrication costs for preliminary estimates by the design engineer. The DTC costing procedures have the flexibility of allowing as much or as little detail as the particular estimate requires. To provide an engineering approach to costing, an effort was made to establish the range and type of alloys expected to be used in the various engine geometries studies, and to establish a means of costing these materials based on current prices, form, and quantity.

A listing of the most common metals, in their various forms, was made by selecting those materials being used in current production and development engines. Through direct contact with materials suppliers and through the utilization of their price listings and current cost estimates, the materials costs were gathered and plotted. The resulting curves reflect base prices. Because base price is somewhat modified by sheet thickness, bar size and purchased weight, several curves were plotted, where appropriate, so that interpolations could be performed. In addition, a tabulation was included on the curve to correct the pricing of small order lots. A typical materials pricing curve is illustrated in Figure 2.2.3-2.

Castings do not fit into any normal manufacturing process such as drilling or milling. In addition, they are subject to many cost variables which result from techniques of pouring, number of cores, thickness of walls, and the like. Therefore, castings are considered to be a unique material form, and have been priced in a manner that is appropriate to that form. Prices of castings, prior to any subsequent machining or processing, were gathered from currently known vendor quotes and costs. The castings selected were representative of various nickel, iron, aluminum, or cobalt alloys, and were representative of the various component forms, such as spinners, turbine rotors, oil pump housings or nozzles. After screening the information in rough plots to establish cost trends, numerical values were assigned to the various factors of casting difficulty.

Complexity factors have also been established for forgings of various materials appropriate for use in gas turbine engines. An effort has been made to establish a means for costing all material previously used in this industry. Additional new alloys and/or fabrication techniques will continue to be incorporated as necessary.

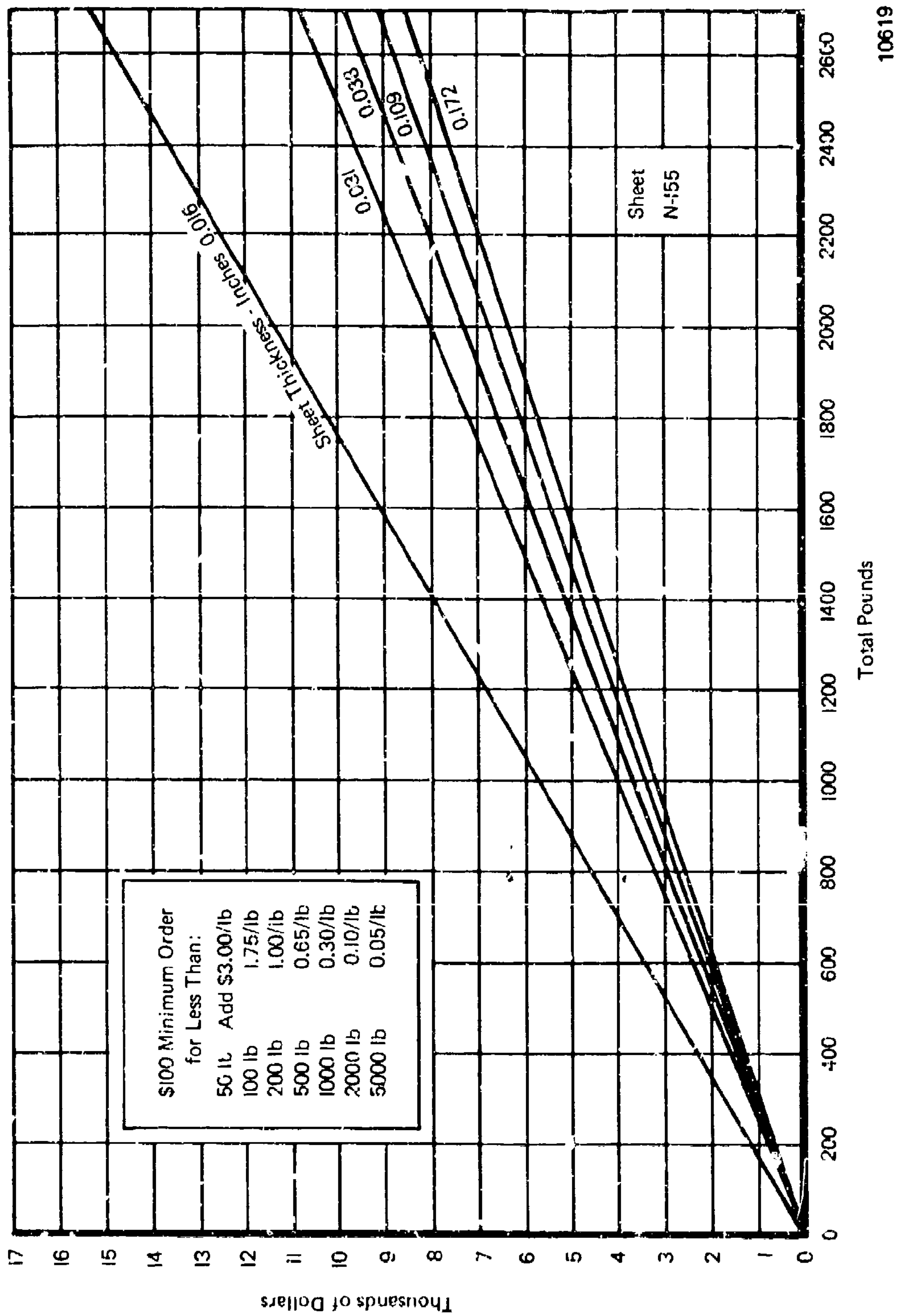



Figure 2.2.3-2. Typical Material Pricing Curve for N-155 Sheet.

A similar effort has been made to establish a means of costing the various manufacturing operations utilized in engine fabrication. It is well known that the ease and effectiveness of manufacturing processes vary broadly with base metal, alloy and heat treat. In order to properly assess the cost effect of these variables, some measure of variation in difficulty had to be established. The machining statistics on the more common methods of manufacturing were gathered from the Machining Data Handbook. Using the data supplied on the materials and methods appropriate to the present program, tabulations were prepared for the various operations using heat treat conditions that were expected to be encountered. A plot of relative material removal rates for single point turning is shown in Figure 2.2.3-3. The ordinate, or the factor of increasing difficulty, is a dimensionless number, which when divided into the removal rate stated on the curve, will correct for the various alloys and heat treats.

Of course, some machining operations do not always fit the situation depicted by idealized curves, so Teledyne CAE's Industrial Engineering Data are used in those cases. This is evidenced in the case of hydrotel airfoil milling (Figure 2.2.3-4) which is peculiar to gas turbine fabrication. Similarly, manufacturing processes such as electrodischarge machining (EDM), electrochemical machining (ECM), welding, brazing, heat treat, surface treat and coatings have been reduced to time elements based on experience. Manufacturing and material costs exist in a dynamic environment and are necessarily revised as changes occur in availability, capability and/or producibility.

Using the procedures and data described, in addition to vendor and manufacturing engineer quotes, design engineering compiled the engine acquisition cost data presented in Section 3.0. These costs were all developed using these assumptions:

- Production quantities of 1,000 engine/year.

- Tooling costs were excluded.

- G & A and profit were excluded.

- 25 percent G & A and 10 percent profit were assumed for system cost payoffs on component cost reduction topics.

In addition to these assumptions, each of the missions analyzed provided typical linearized weighting factors (for each delivered aircraft) for improvements in weight and SFC as related to dollar savings. These dollar equivalents were used to provide system cost payoffs for the component cost reduction topics.

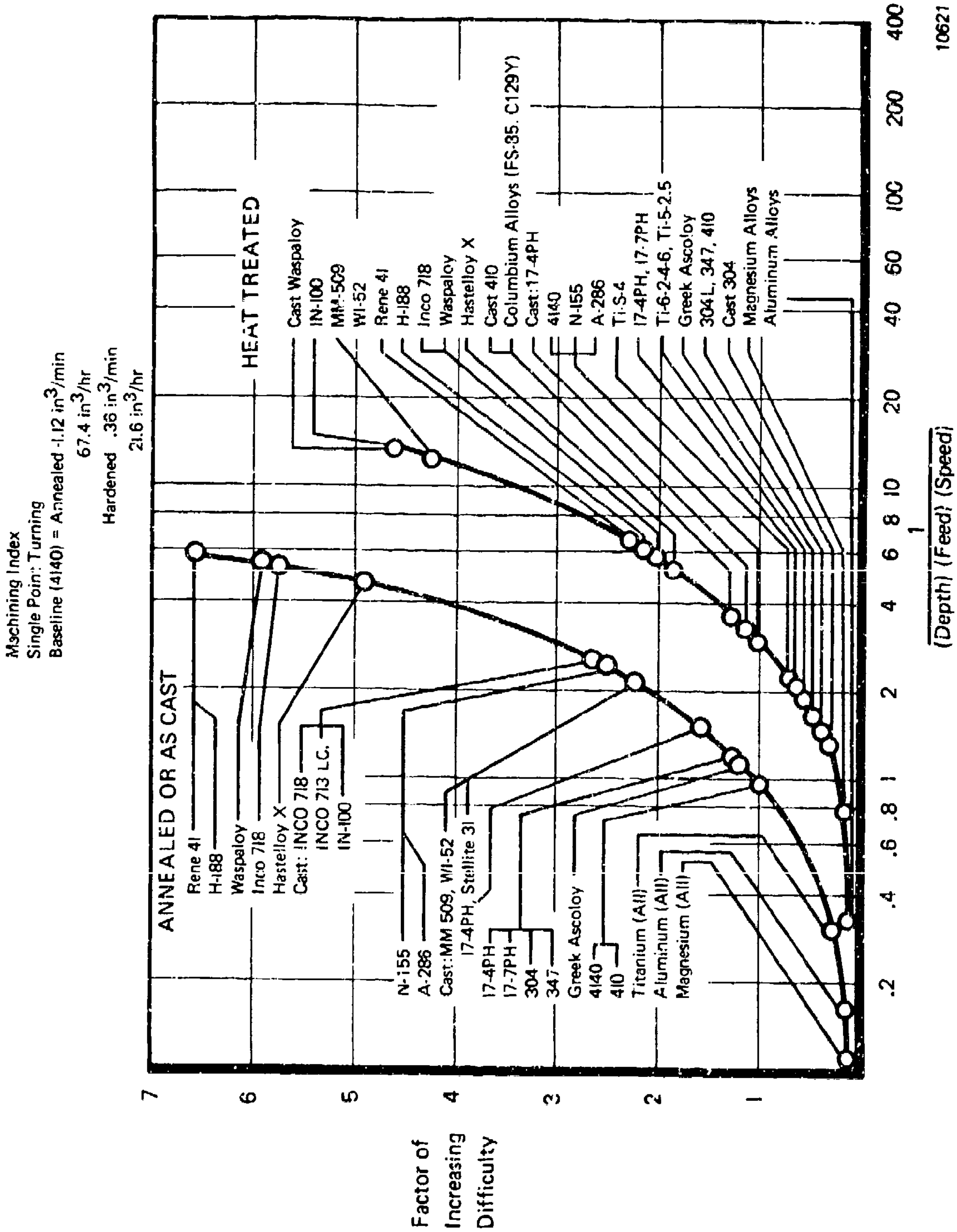



Figure 2.2.3-3. Typical Machining Index for Single Point Turning.

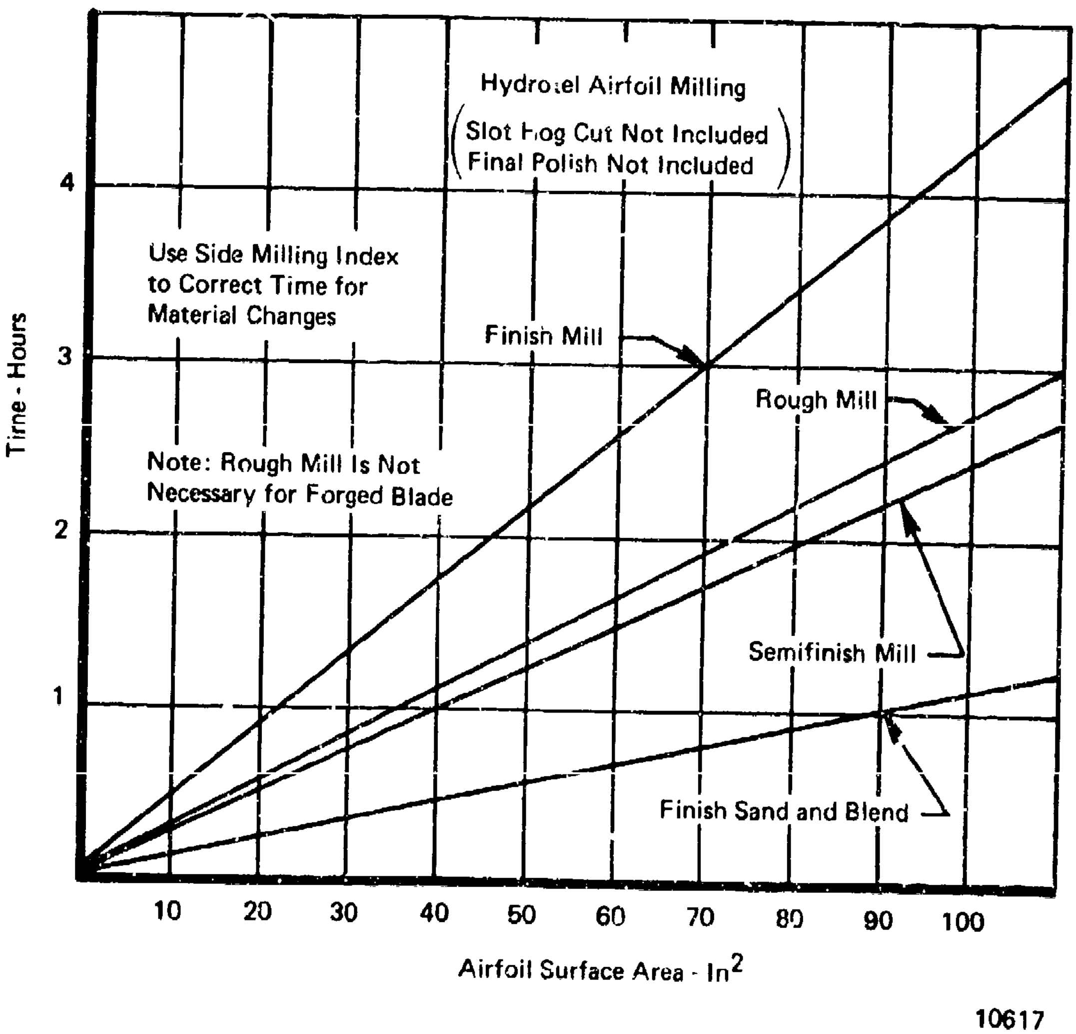



Figure 2.2.3-4. Hydrotel Airfoil Milling.

## 2.2.4 Operation and Support Costs

The APSI program provided a means to challenge the trends in total costs of engine ownership, and the cost of operation and support was also addressed.

Two specific areas of operation and support were evaluated by Teledyne CAE during the APL APSI program. These are:

Operational costs of the engine fleet (consists principally of fuel and lubricant consumption).

Maintenance support costs including manhours, material (replacement part) costs, and inventory management costs.

During APSI, the airframer's studies of engine/air vehicle interaction emphasized that fuel consumption should properly be addressed as an aircraft/engine interactive cost. Maintenance support costs, on the other hand, are primarily determined by the engine designer in all areas with the exception of "installed-engine-accessibility".

The 455 series addresses the maintenance cost problem therefore in these important aspects:

The development program for the JTDE and the "delta" development will field a family of engines that are relatively mature in terms of verified reliability and durability.

The engine family concept reduces the number of parts and spreads the inventory management cost ($120/part-type/year) across the highly common engine types.

The inherent simplicity of the 455 series will reduce maintenance training, documentation and hands-on repair tasks.

The low acquisition cost reduces lifetime engine and component replacement costs.

## 2.2.5 Performance (Interactive) Cost

The Teledyne CAE subcontracted (APL APSI program-sponsored) aircraft engine studies addressed the task of selecting optimum engine/aircraft configurations. An important spin-off of these studies was the recognition that an engine's form and function have significant impact on the development, acquisition and lifetime ownership costs of the air vehicle.

These results are not remarkable when the cost of aircraft materials is considered, but derived values of their slopes (e.g., air vehicle cost versus engine weight/SFC) become very significant for smaller aircraft and engines.

The initial effect impacts airframe acquisition cost as a function of engine weight and SFC. However, these initial effects of engine performance are carried through the life of the air vehicle. For example:

> Airframe lifetime maintenance costs are partially a function of aircraft weight, which is initially sized by the SFC and weight of available engines.
>
> Mission and lifetime fuel consumption are affected by engine SFC and engine thrust-to-weight, because both SFC and the latter ratio "size" the airframe and its maximum fuel capacity.

The APSI program addressed both SFC and thrust-to-weight ratio to develop engines providing significant improvements in these two airframe-interacting parameters, as well as on their trade-off with engine acquisition costs.

# Section 3.0
# APSI Cost Reduction

# SECTION 3.0 - APSI COST REDUCTION

## 3.1 Background

Approaches for addressing the performance, cost, and design-life of propulsion systems require carefully directed use of an iterative process. This process begins with definition of potential applications and their broad range of performance requirements. It follows with a definition of aerodynamic and thermodynamic considerations which, in turn, establish cross section, size, volume, and mechanical characteristics of the proposed engine. The first baseline design then provides the reference for parametric evaluations (with respect to requirements), sensitivity determinations, and cost investigation tradeoffs.

The APSI series of engines extends the basic design approach of the Advanced Turbine Engine Gas Generator (ATEGG), which results from the above described iterative process. In ATEGG, Teledyne CAE considered a range of prospective engine-derivative applications as listed in Figure 3.1-1. The application considerations provided direction to pursue the underlying objective of advancing the technology level of engines in the thrust class of interest. Because of the long period of expected development and therefore the likelihood of changes in specific performance requirements, an additional requirement was identified. This requirement was that the gas generator be highly flexible and compatible with a wide range of future performance requirements. These considerations established the set of performance requirements.

The qualitative objectives for the ATEGG required a series of quantitative evaluations and commitments. For example, it was assumed that prospective missions would require good SFC, for either life cycle cost improvement or adequate loiter time and range. Hence, at least some 455 engine derivatives would require very low SFC's.

Compressor configurations were therefore evaluated to select a design that offered the desired pressure ratio and efficiency at the least cost and weight. Seventeen candidate configurations were thoroughly examined prior to final selection. Comparable evaluations were conducted to optimize other components. Contact with USAF/APL Plans and Programs Office further substantiated the future need for engines in this class. Sufficient requirements data were made available to validate the design choices.

| TPO NO. | (SCP) | APPLICATION | THRUST | SFC REQUIREMENT | BYPASS RATIO | TIT-°F |
|---|---|---|---|---|---|---|
| - | - | TA-3<br>T-38 & 39 Replacement | 4000 | Low | High | High |
| - | - | TA-4 Transport/Bomber/Trainer | 3000 | Low | High | Low |
| #11 | (15) | Low Altitude Tactical<br>Reconnaissance Unmanned | 5000 | Low | Low | High |
| #12 | (14) | Quiet Aircraft (FAC-X) | 4000-5000 | Low | High | Interm. |
| #11 | (12) | Remotely Piloted Vehicle Air to Ground System | | | | |
| | | 1. Low Cost Expendable | 500-1000 | Not Significant | 0 | Uncooled |
| | | 2. Recoverable | 1000-3000 | Moderate to Low | Low | High |
| | | 3. Recoverable | 2000-3000 | Moderate to Low | Low | High |
| #11 | (119) | RPV's | | | | |
| | | 1. Reconnaissance Long Duration | 3000 | Low | Interm. | Interm. |
| | | 2. Reconnaissance Deep Penetration | 3000 | Moderate | Interm. | Interm. |
| | | 3. Air-To-Air Combat | 3000 | Moderate | Low | High |
| #11 | (95) | Full Scale Maneuvering Target (FSMT) (Recoverable) | 4000-5000 | Moderate | 0 | Uncooled |
| #11 | (94) | High Altitude Supersonic Target (HAST) | | Moderate | 0 | High |
| - | - | ZBQM-90 | 3000 | Moderate | 0 | High |
| - | - | Microfighter | 5000-8000 | Moderate | Low | Interm. |
| #11 | (15) | ECM Support Mission (Expendable) | | Not Significant | 0 | Uncooled |

(1) Cruise
(2) Maximum

19470

Figure 3.1-1. Derivative Engine Applications.

A specific dollar cost target was not established for ATEGG or its possible derivatives. However, a second underlying objective was to establish the lowest inherent cost for a given performance objective. Tradeoff studies were conducted to this end. The expected result is that the best cost pay-off will result from simplifying the flowpath, eliminating stages, and emphasizing the use of near-size forgings and castings to minimize the ratio of raw material weight to fly-away weight (or Fly/Buy Ratio).

In summary, the challenge to mechanical design for a man-rated ATEGG derivative resulted from consideration of performance, long range cost, and optimum utilization and return from development engineering effort. The response evolved as a man-rated engine, having a highly-loaded axi-centrifugal compressor and single-stage cooled turbine.

## 3.2 Baseline Engines

A single model was selected as the baseline engine for the APSI cost studies. It is a potentially optimum propulsion system for a low level transonic mission, combining a relatively high fan pressure ratio with a moderate bypass ratio. The design also envisions a relatively high tip speed and high fan stage turbine loadings to limit the number of low pressure stages. This provides a compact, high performance design. This design, with a minimum number of components, provides an inherently low cost baseline engine against which all cost reduction items have been evaluated.

For cost reduction purposes, the engine was divided into ten component assemblies which defined the cost groups. The cost groups were further broken down into the components which make up a given cost group. This procedure ensured that all the necessary engine hardware was accounted for and that none were overshadowed by relative high cost items of other components which represent the 80 percent/20 percent group. The cost group summary sheets (sample illustrated in Figure 3.2-1) enumerate the parts on the 80 percent/20 percent factor where 20 percent of the parts represent 80 percent of the cost of each individual group. The remaining parts are the miscellaneous parts which represent 20 percent of the group cost and were not cost itemized. Each of the cost parts has been estimated in detail. An example of the detail part cost work sheet is shown in Figure 3.2-2. The detail cost worksheets were used to provide the cost inputs for the group summary sheets. These group summary sheets, in turn, were used to compile the engine cost estimate (Figure 3.2-3). Parts costs were then used to provide tradeoff comparisons for component cost reduction items as applicable.

## 3.3 Scaled Engine Costs

The baseline engine selection was made using the same philosophy applied in the selection of the gas generator for the ATEGG program. The engine

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1330 LASKEY ROAD · TOLEDO OHIO 43612
Turbine Engines Ideas With Power

DESIGN-TO-COST ANALYSIS (TYPE II)
SEE ESP NO. ______ FOR INSTRUCTIONS
ENGINE GROUP COST ESTIMATES

ENGINE MODEL NO. ______
ENGINE GROUP NO. ______
ENGINE GROUP NAME ______
PAGE ____ OF ____ DESIGN-TO-COST ESTIMATE NO. ______

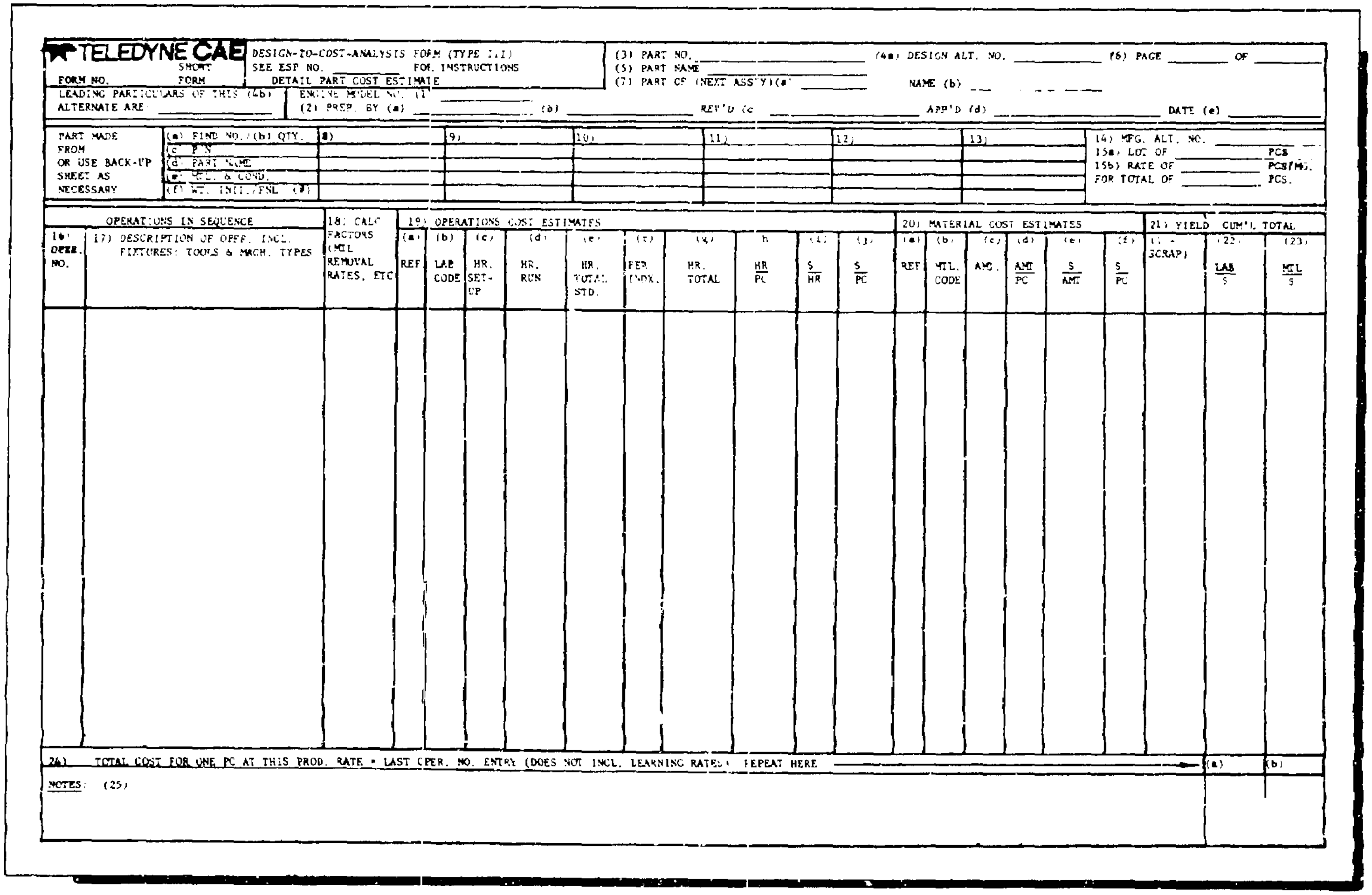
PREPARED BY: ______ REVIEWED BY: ______ APPROVED BY: ______ DATE APPROVED: ______

| DETAIL PARTS (IN ORDER OF TOTAL COST)* | | | | | LABOR \$ | | | | | | | MATERIAL \$ | | | | | | | | | | |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| | | | | | BY LABOR CODES (GENERAL)* | | | | | | | BY MATERIAL CODES | | | | | | | | | | |
| FIND NO. | DETAIL P/N | ALT. NO. DES | ALT. NO. MFG. | PART NAME (AS SHOWN ON DRAWING) | | | | | | MISC. | TOTAL \$ LABOR | | | | | | MISC. | TOTAL \$ MATERIAL | TOTAL \$ LABOR \$ MATERIAL | % RANK ** | NOTES |
| | | | | | | | | | | | | | | | | | | | | | |
| TOTALS FOR THIS ENGINE GROUP | | | | | | | | | | | | | | | | | | | | 100% | |

* MAY BE OMITTED WHEN EDP METHODS ARE NOT USED.
** IF SHOWN ON ENGINE PARTS LIST.

FORM T-703A 2/76

Figure 3.2-1. Sample: Cost Group Summary Sheet.



TELEDYNE CAE SHORT FORM
FORM NO.
DESIGN-TO-COST-ANALYSIS FORM (TYPE 1.1)
SEE ESP NO. FOR INSTRUCTIONS
DETAIL PART COST ESTIMATE
(3) PART NO.
(4a) DESIGN ALT. NO.
(6) PAGE OF
(5) PART NAME
(7) PART OF (NEXT ASS'Y)(a)
NAME (b)
LEADING PARTICULARS OF THIS (4b) ALTERNATE ARE
ENGINE MODEL NO. (1)
(2) PREP. BY (a) (b) REV'D (c APP'D (d) DATE (e)
PART MADE FROM OR USE BACK-UP SHEET AS NECESSARY
(a) FIND NO./(b) QTY.
(c) P/N
(d) PART NAME
(e) MTL. & COND.
(f) WT. INIT./FNL (g)
8) 9) 10) 11) 12) 13)
14) MFG. ALT. NO.
15a) LOT OF PCS
15b) RATE OF PCS/MO.
FOR TOTAL OF PCS.

| OPERATIONS IN SEQUENCE | | 18) CALC FACTORS (MTL REMOVAL RATES, ETC | 19) OPERATIONS COST ESTIMATES | | | | | | | | | | | 20) MATERIAL COST ESTIMATES | | | | | | 21) YIELD | CUM'L TOTAL | |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| 16) OPER. NO. | 17) DESCRIPTION OF OPER. INCL. FIXTURES: TOOLS & MACH. TYPES | | (a) REF | (b) LAB CODE | (c) HR. SET-UP | (d) HR. RUN | (e) HR. TOTAL STD. | (f) PER INDX. | (g) HR. TOTAL | h HR/PC | (i) \$/HR | (j) \$/PC | (a) REF | (b) MTL. CODE | (c) AMT. | (d) AMT/PC | (e) \$/AMT | (f) \$/PC | (1 - SCRAP) | (22) LAB/\$ | (23) MTL/\$ |

24) TOTAL COST FOR ONE PC AT THIS PROD. RATE • LAST OPER. NO. ENTRY (DOES NOT INCL. LEARNING RATE) REPEAT HERE → (a) (b)
NOTES: (25)

Figure 3.2-2. Sample: Detail Part Cost Worksheet.

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1330 LASKEY ROAD TOLEDO OHIO 43612
Turbine Engines Ideas With Power

DESIGN-TO-COST ANALYSIS FORM (TYPE 1)
SEE ESP. FOR INSTRUCTIONS
ENGINE MODEL COST ESTIMATE

ENGINE MODEL NO. ____
ENGINE DESIGNATION ____
PROGRAM ____

ENGINE DRAWING NO. ____ REVISION NO ____ SPECIFICATION NO. ____ REV. ____ STATUS OF ESTIMATE ____
PREL. ____ QT ____ PROD. ____ PREPARED BY ____, ____ REVIEWED BY ____, ____
APPROVED BY ____, ____ DATE ____ SHEET ____ OF ____

| ENGINE COST GROUPS | P/N OR DESIGNATION | PART NAME OR DESCRIPTION | TOTAL LABOR $ | TOTAL MATERIAL $ | TOTAL (ACT $ LABOR + MATERIALS) | % RANK OF TOTAL COSTS | TARGETED OR COMMITTED COST $ | VARIANCE + OR - (%)(3) | NOTE NO. |
|---|---|---|---|---|---|---|---|---|---|
| BARE ENGINE | | | | | | | | | |
| | | BARE ENGINE ASS'Y & BUILD-UP | | | | | | | |
| | | TOTALS FOR BARE ENGINE | | | | | | | |
| MATERIAL MISC. (1) | | HARDWARE | | | | | | | |
| | | BULK MATERIAL | | | | | | | |
| | | TOTAL MISC. MATERIAL | | | | | | | |
| CONTROLS & ACCESSORIES | | | | | | | | | |
| | | TOTAL CONTROLS & ACCESSORIES | | | | | | | |
| ENGINE SYSTEM | | ASSEMBLY | | | | | | | |
| | | ASSEMBLY INSPECTION | | | | | | | |
| | | TEST (1) | | | | | | | |
| | | (2) | | | | | | | |
| | | (3) | | | | | | | |
| | | (4) | | | | | | | |
| | | FINAL ASSEMBLY | | | | | | | |
| | | FINAL INSPECTION | | | | | | | |
| | | PACKING & SHIPPING | | | | | | | |
| | | (OTHER) | | | | | | | |
| | | TOTAL FOR ENGINE SYSTEM | | | | | | | |
| TOTAL COST TO SHIP THIS MODEL NO. ENGINE (2) | | | | | | 100% | | | |

NOTES:
(1) NOT INCLUDED IN DETAIL PART ESTIMATES.
(ATTACHING HARDWARE, LUBRICANTS, SHIPPED TEST PARTS FOR TEST CONFIGURATIONS)
(2) EXCLUSIVE OF G&A AND/OR FEE, = "BASELINE COST" IF LEARNING IS USED.
(3) VARIANCE (%) = (TARGET-ACTUAL/TARGET) x 100.
(4) ESCALATION BASED ON ____.

LEARNING CURVE ADJUSTMENT (ATTACH ANALYSIS)
TOTAL PRODUCTION QUANTITY
TOTAL NO. OF MONTHS OR LOTS
DIFF. INCREASE OR (DECREASE) FROM BASELINE FOR LOT NO.'S (%) IS.

| NO./LOT NO. | | | | | | | | | | | | |
|---|---|---|---|---|---|---|---|---|---|---|---|---|
| % DIFF. | | | | | | | | | | | | |

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Figure 3.2-3. Sample: Engine Model Cost Estimate.

components had to be extremely versatile and readily scalable. To develop cost data for scaled versions of the baseline engine, a twice-thrust scale and a one-half thrust scale were selected.

## 3.4 System Cost Reduction

### 3.4.1 Summary

During the APSI program, Teledyne CAE addressed a number of system cost considerations in optimizing the engines for aircraft applications. To assist in this task, Teledyne CAE funded two airframer studies that defined optimum aircraft/engine combinations for a future undergraduate pilot trainer or UPT (representative manned mission), and for a multi-mission remotely piloted vehicle or MMRPV (representative unmanned mission). The UPT and MMRPV missions were selected as benchmarks because:

> The UPT bounds the "high end" of the mission spectrum in terms of annual utilization and total life.

> The MMRPV has good prospects for near term development and it bounds the low-end of the mission spectrum.

The mission studies highlighted the influence of engine optimization on an air vehicle's total lifetime cost of ownership. The studies suggest, for example, that the interactive (engine impact on airframe design) costs may equal or exceed the (more generally recognized) explicit costs of engine ownership during the air vehicle lifetime. The studies also indicate that only one to three percent of the total cost of engine ownership is available for development test and evaluation, which emphasizes the leverage of engine development on lifetime costs.

The foregoing considerations led to the selection of cost-optimum design to:

> Combine the ATEGG with the APSI-sponsored fan turbine and company-developed fan to derive a turbofan engine parent to a variety of derivative turbofan engines.

> Serve as a realistic, engine-operating environment for evaluating the systems payoff of reduced-cost engine components.

The selected engine provides the best baseline for low cost derivatives because it is inherently low-cost in its own right. Notable features are:

The mechanical arrangement uses a minimum of parts.

The fan and HP compressor are designed for casting to reduce initial manufacturing and lifetime replacement costs.

The fan turbine is designed by cost drivers.

The counter-rotating LP shaft reduces gyroscopic loads on both engine and airframe, reduces the stress (and cost) of mounting provisions, and simplifies (and reduces cost of) the low pressure spool turbine inlet nozzle.

The selected engine can be tailored, by scaling and/or component modification, into a low cost powerplant for a wide range of future requirements. Its span of applications includes the unmanned MMRPV through the manned and heavily used UPT. The potential cost leverage of APSI technology is exemplified by the data shown in Figure 3.4.2.

The maximum payoff from APSI technology will require careful matching of derivative engine designs to their potential aircraft applications. To that end, Teledyne CAE has evolved a method for evaluating system costs (Section 2.2.3). This approach was used to evaluate the Teledyne CAE MMRPV and UPT candidates. It was also used to evaluate candidate engine components that show good prospects for further cost reduction in several of the baseline series of engines.

The results demonstrate that fully discounted lifetime savings from the first two APSI-derivative development programs would "payback" 20 times the Military Services' investment in exploiting technology to reduce engine systems cost.

### 3.4.2 Mission Definitions

The mission spectrum for the selected derivative engine spans a wide range of future military aircraft requirements. Missions, utilization rates and total life cycles will vary, and will therefore require individual evaluation of systems cost and benefits. An illustration of their variety and diversity is shown in Figure 3.4.2, which tabulates several aircraft parameters that impact life cycle cost.

The fleet size, utilization and life span estimates in Figure 3.4.2 are based on Teledyne CAE records for comparable present day aircraft such as the T-37 (J69-T-25 engine) and special purpose aircraft (SPA's) that incorporate the J69-T-29, J69-T-41, J69-T-406, and J100-CA-100. Also, best utilization rates are shown for manned aircraft because they represent good DOD cost reducing strategy.

| Number of Aircraft | Number of Engines/AC | Annual Utilization Aircraft Hr/Yr | Projected Peak and (Total) Life Cycle Years | Projected Total Fleet Lifetime Engine Operating Hours (Millions) | Prospective Aircraft Type |
|---|---|---|---|---|---|
| 700 to 1100 | 2 | 720 | 15 (25) | 24 - 39 | UPT |
| 300 to 600 | 2 | 480 | 15 (25) | 7 - 15 | Liaison |
| 300 to 600 | 2 | 360 | 10 (20) | 3.5 - 7.0 | Deck-Launched Interceptor |
| 200 to 400 | 1 | 480 | 15 (25) | 2 - 5.0 | Forward Air Controller |
| 500 to 900 | 1 | 60 | 10 (20) | 1 - 1.5 | RPV's MM ASF CAS ASW |

Figure 3.4.2. Aircraft Parameters Affecting APSI Cost.

The MMRPV and UPT occupy places of special interest in the prospective applications list because they both have good prospects for use and because they tend to bound the mission spectrum. The UPT mission, over its lifetime of 24-39 million engine hours, will benefit from the performance (particularly SFC), low initial cost, and maintenance-cost-reducing simplicity of the selected turbofan.

Conversely, the MMRPV will place a higher value on the low acquisition cost, high thrust-to-weight, and survivability features of the selected turbofan.

3.4.3 UPT Mission and Engine Model Selection

During the APL APSI contract, Teledyne CAE selected an airfram r to study future manned-aircraft applications of APSI derivative engines. The USAF Undergraduate Pilot Trainer (UPT) was selected for this purpose because it has near-term prospects, has the highest potential annual utilization during its 15 peak flying years and has a total life of about 25 years.

The airframer identified 12 UPT training mission profiles that Teledyne CAE has combined into a "composite mission" composed of time-weighted segments. A "segment" represents a specific engine operating point in terms of altitude, Mach number and power setting, as illustrated in Figure 3.4.3-1.

The composite mission provides a reference for establishing a number of cost parameters significant to life cycle cost reduction efforts. For example, the UPT engine will spend about 20 percent of its operating hours at idle power and sea level static conditions, which emphasizes the need for good, reduced-power SFC.

Companion engine/mission cost considerations are the annual and cumulative engine operating hours that result from fleet size, utilization rate and expected time in inventory. A number of Teledyne CAE data sources were reviewed to construct a projected life cycle for the UPT application. These results (and their source data) are noteworthy in that:

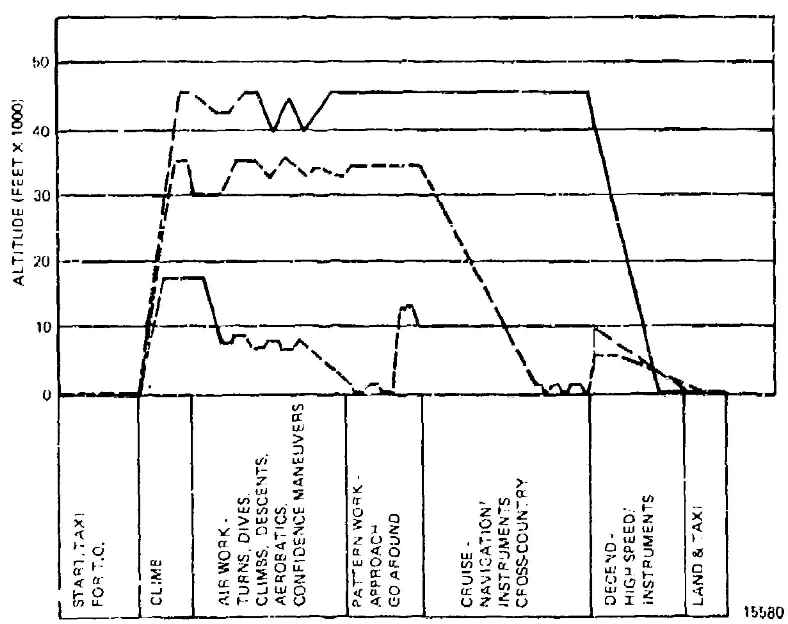


Figure 3.4.3-1. UPT Composite Mission Profile (Engine Operating Hours).

The total life of a UPT can exceed 25 years, while its peak life of 15 years will account for about 78 percent of engine useage.

The UPT's expected utilization of 60 hours a month will make it a cost-effective trainer aircraft.

### 3.4.4 MMRPV Engine Selection

During the APL APSI contract, Teledyne CAE selected an airframer to perform system studies of a future unmanned derivative application. The Air Force's Multi-Mission Remotely Piloted Vehicle (MMRPV) was chosen for this purpose, and the airframer identified a number of air vehicle and engine candidate configurations. A mission optimized MMRPV was then selected, and engine configuration was evolved to fulfill its requirements.

An MMRPV engine life cycle was also postulated from Teledyne CAE's data banks on drones, special purpose aircraft and RPV's. A typical RPV engine life cycle is shown in Figure 3.4.4, which graphs annual and cumulative engine hours versus life cycle years. The plotted value of one million hours for MMRPV engine life tends to be conservative because of the increasing number of uses identified for RPV's. A probable upper limit is likely to approach seven million engine life hours (typical of a tactical aircraft), as RPV's tend to fill tactical roles and missions.

### 3.4.5 System Cost Comparisons

A principal objective of the APSI is reduction of DOD real engine-ownership costs. During the APL APSI program, these costs were explored in a number of studies. Two studies of airframe-engine cost interaction were supported by airframe-contractor efforts. Other studies involved Teledyne CAE research. Two of the many conclusions deserve particular mention:

The real costs of engine ownership must include consideration of its impact on air vehicle cost.

Only one to three percent of the total cost of engine ownership is devoted to engine development - but savings from properly directed development engineering can exceed 20 times the development investment.

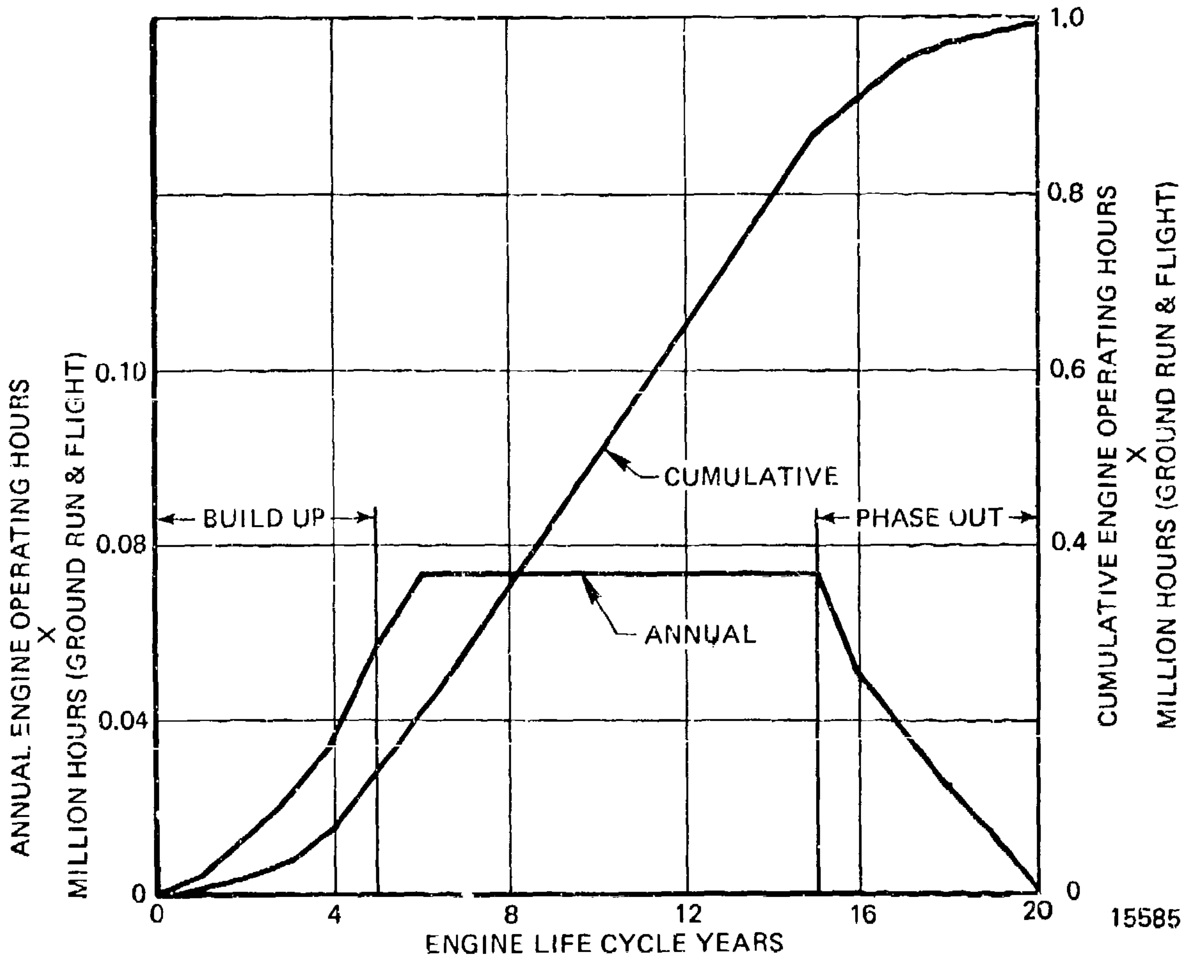


Figure 3.4.4. Postulated RPV Cumulative and Annual Engine Operating Hours (Life Cycle).

These conclusions were substantiated in a comparison involving two derivative engines and their current-technology counterparts. For comparison purposes, one ATEGG derivative was compared to a high bypass ratio engine for the UPT mission; and another was compared to a turbojet engine for the MMRPV mission. The current engines were selected on the basis of the closest fit available for the applications. Also, best estimates of their characteristics were used.

Engine parameters of interest are those with large, first-order, cost impact. These are listed for both the APSI derivatives and their current counterparts in Figure 3.4.5-1. The ownership parameters and comments relating its use in estimating engine ownership cost impact are listed in Figure 3.4.5-2.

These comparisons were made on the basis of individual, linearized estimates; the important effects of multi-parameter sensitivity were beyond the scope of current tasks. Teledyne CAE recommends that mutli-parameter effects be addressed as an element of the Design-to-Life-Cycle-Cost model described in Section 5.0 of this report.

| ENGINE PARAMETER/ CHARACTERISTIC | ENGINE OWNERSHIP COST AREA IMPACT | PARAMETER VALUES AND JTDE DELTAS | | | | | |
|---|---|---|---|---|---|---|---|
| | | MMRPV | | | UPT | | |
| | | "CURRENT" ENGINE | | DELTA PER ENGINE | "CURRENT" ENGINE | | DELTA PER ENGINE |
| Engine Weight (Lb) | • Airframe Development Acquisition and Lifetime Maintenance | 100% | 85.4% | -14.6% | 100% | 59.8% | 40.2% |
| Engine Specific Fuel Consumption (SFC) SLS, MIL PWR | • Airframe Development Acquisition and Lifetime Maintenance (Sizes Airframe and Weight) | | | C L A S S | I F I E D | | |
| Engine Average Specific Fuel Consumption in Referenced Mission ($\overline{SFC}$) | Lifetime Aircraft Fuel Consumption | | | C L A S S | I F I E D | | |
| Engine Thrust to Weight Ratio | • Aircraft Fuel Consumption, Weight and Form Factors + All of Above When Used as Alternate Sizing Reference | | | C L A S S | I F I E D | | |
| Complexity (No. of Separate Parts) | • Initial Sparing Cost ✓ Lifetime Stock Management Cost | 100% | 90% | -10% | 100% | 85% | -15% |
| Engine Maintenance Index | Man/FH of Direct Maintenance (Engines Only) | 100% | 83.8% | -16.2% | 100% | 90.2% | -9.8% |

Figure 3.4.5-1. APSI-Derivative Engines Compared with Closest Available Current Technology for UPT and MMRPV Missions.

| PARAMETER | | COMMENT |
|---|---|---|
| (1) | FUEL | Current Military Price = $0.39 |
| (1.1) | JP-4 | Per Gallon of JP-4 at Nozzle |
| (1.2) | Booster Launch/lb of Engine | Can also vary with SFC (MMRPV Only) |
| (2) | AIRFRAME DEVELOPMENT | |
| | •UPT | |
| (2.1) | ••Per lb of Engine | Relatively Insensitive to Fleet |
| (2.2) | ••Per point (0.01) SFC | Size |
| | •MMRPV | |
| (2.3) | ••Per lb of Engine | Implicit in Vehicle Production |
| (2.4) | ••Per point (0.01) SFC | Cost (see Figure 6.5-3) |
| (3) | AIRFRAME ACQUISITION | |
| | •UPT | |
| (3.1) | ••Per lb of Engine | Based on "Nominal" Fleet |
| (3.2) | ••Per point (0.01) SFC | |
| | •MMRPV | |
| (3.3) | ••Per lb of Engine | Based on "Nominal" Fleet |
| (3.4) | ••Per point (0.01) SFC | |
| (4) | AIRFRAME MAINTENANCE | NOTE: Following should also be sensitive to SFC improvements |
| | •UPT | |
| (4.1) | ••Per lb of Engine | |
| | | No "credit" for residual airframe weight savings |
| | •MMRPV | |
| (4.2) | ••Per lb of Engine | Best (most conservative) estimate |
| (5) | ENGINE MAINTENANCE | |
| (5.1) | •Direct Labor | Estimate Range = $5-17/Hour |
| (5.2) | •Material | Estimate Range = 0.04 - 0.10%/Year of Engine Acquisition Cost ($C_A$) |

Figure 3.4.5-2. Parameter Values for Estimating Derivative Engine System Cost Savings.

# Section 4.0
# Cost Reduction Analyses

# SECTION 4.0 - COST REDUCTION ANALYSES

## 4.1 AERODYNAMIC TRADEOFFS

The cost reduction analyses were conducted in two stages: analysis of aerodynamic design parameters influencing cost; analysis of specific cost reduction topics for design and manufacturing changes to reduce cost, reduce weight, or improve performance yielding a minimum system cost.

Application of aerodynamic design tradeoffs in the selection and optimization of component designs can have a significant system cost impact.

### 4.1.1 Reduced Cost Parameters for Compressors

This analysis has the objective of extending cost reduction by definition of cost related parameters for compressors and to define generalized curves or relationships representing a best estimate of the effect of these parameters on aerodynamic performance for a single-stage compressor. Stage matching effects were not considered. Only design point performance changes were considered.

Estimates are presented herein on the effects of tip clearance/span ratio, blade surface finish-to-chord ratio and rotor solidity ratio, corrected flow and efficiency. Also discussed are the effects of rotor aspect ratio and blade edge thickness. For centrifugal stages, the effect of tip clearance on pressure ratio, airflow and efficiency is indicated.

Table 4.1.1-I presents an example of the performance degradation of a typical Teledyne CAE axial compressor first stage due to the effects of the above parameters.

#### 4.1.1.1 Effect of Tip Clearance/Span Ratio for an Axial Stage

The data for tip clearance effects were obtained from References 1 through 5. The performance degradation model was developed assuming that no performance changes occur when the tip clearance is less than or equal to the boundary layer displacement thickness as determined by the design annulus wall blockage.

Effect on Efficiency - Efficiency degradation versus tip clearance/span ratio data is shown in Figure 4.1.1-1. For the selected variation of polytropic efficiency, derated stage adiabatic efficiency is given by:

TABLE 4.1.1-I

PERFORMANCE DEGRADATION EXAMPLE

| CASE | CONDITION | PRESSURE RATIO % of Ref. | CORRECTED FLOW % | ADIABATIC EFFICIENCY $\Delta\eta$ (%) | POLYTROPIC EFFICIENCY $\Delta\eta$ (%) |
|---|---|---|---|---|---|
| 1 | Reference Design | 100 | 100 | Baseline | Baseline |
| 2 | Same as Case 1, but with doubled rotor tip clearance | 98.25 | 99.5 | 1.31 | 1.25 |
| 3 | Same as Case 2, but with surface finish increased from 63 to 80 | 97.85 | 98.8 | 2.18 | 2.07 |
| 4 | Same as Case 2, but with surface finish increased from 63 to 125 | 96.8 | 97.1 | 4.48 | 4.22 |
| 5 | Same as Case 4, but with rotor leading edge thickness doubled (assumes limited redesign and/or up-scale) | 96.8 | 97.1 | 6.83 | --- |

$$\eta ad' = \frac{PR'^{(\gamma-1)/\gamma} - 1.}{e^{\left(\frac{\gamma-1}{\gamma} \frac{\ln PR'}{\eta p'}\right)} - 1.}$$

Where

$$\eta p' = \eta p, \text{ref}^* (1. - 1.5801\ (b-c)),$$

for $b > c$,

$$\eta p, \text{ref} = \left(\frac{\gamma-1}{\gamma}\right) \frac{\ln PR_{ref}}{\ln\left(1. + \frac{PR_{ref}^{(\gamma-1)/\gamma} - 1.}{\eta}\right)},$$

and

$\eta ad$ = stage adiabatic efficiency

$\eta p$ = stage polytropic efficiency

PR = stage pressure ratio

$\gamma$ = ratio of specific heats for air at average temperature between stage inlet and exit

b = $\Delta/h$

c = $\delta/h$

$\Delta$ = tip clearance, inches

$\delta$ = tip boundary layer displacement thickness, inches

h = average blade span, inches

Superscript ' refers to derated performance at $b > c$.

Superscript ref refers to reference performance at $b \leq c$.

The model assumes that stage polytropic efficiency is reduced as tip clearance increases beyond the reference boundary layer displacement thickness. Polytropic efficiency reduction varies linearly with clearance/span increase according to the selected fit of empirical data shown on Figure 4.1.1-1 with the exception that the line shown on this figure is translated right or left such that $\eta p'/\eta p$, ref = 1.0 for $\Delta/h = \delta/h$.

OPEN SYMBOLS - ADIABATIC EFFICIENCY
CLOSED SYMBOLS - POLYTROPIC EFFICIENCY

| | | | PRSA |
|---|---|---|---|
| ○ | ASME 67-FE-16, MELLOW & STRONG, SINGLE STAGE | | ~1.14 |
| □ | ASME 60-WA-17, | J79; $\eta$max at $\Delta/h$ = .01; 17 STAGE | 1.16 |
| ◇ | | J79; $\eta$max at $\Delta/h$ = 0. ; STAGE | 1.16 |
| ◊ | | T58; $\eta$max at $\Delta/h$ = .01; 8 STAGE | 1.2865 |
| ◺ | | T58; $\eta$max at $\Delta/h$ = 0. ; STAGE | 1.2865 |
| △ | REPT. R-4; LINDSEY (A-S MAMBA) = .0265; 3 STAGE | | ~1.17 |
| ◗ | AFAPL-TR-72-8 - ADVANCED SEAL TECHNOLOGY | | — |

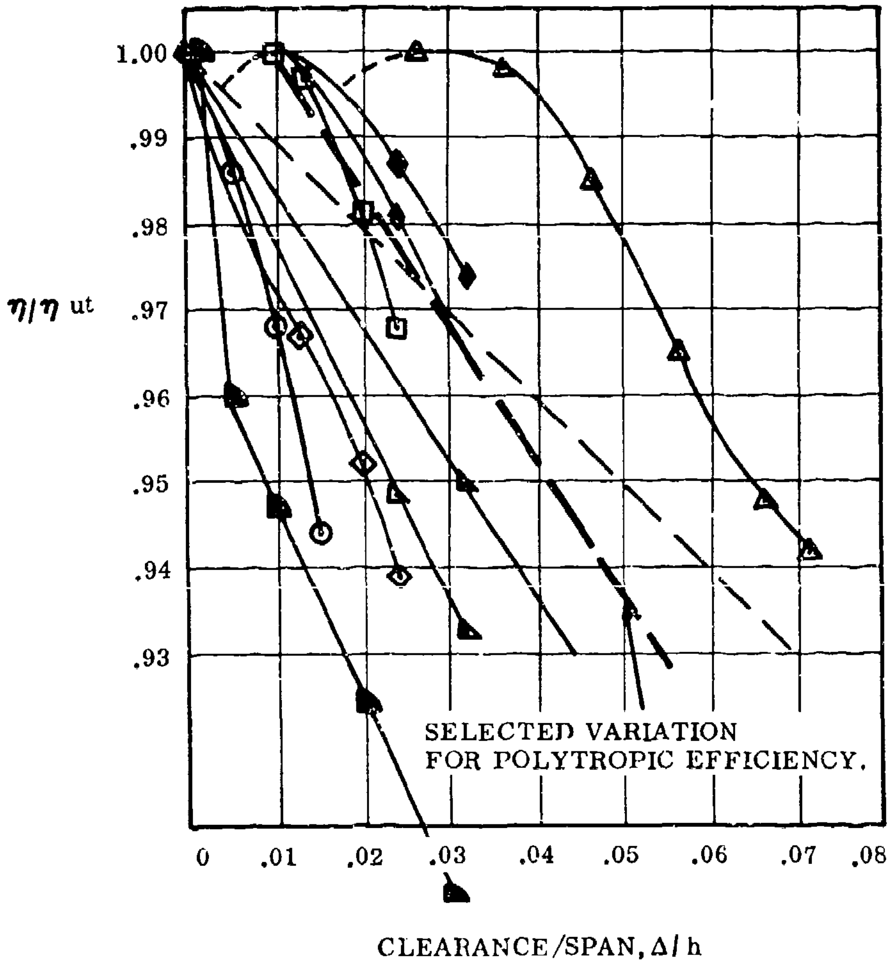


Figure 4.1.1-1. Effect of Tip Clearance on Axial Compressor Efficiency.

Application of this model is limited by the following considerations:

1. It is deduced primarily from multi-stage data.

2. Stage average pressure ratios for the data compressors, at from 1.14 to 1.29, are considerably lower than current practice.

3. There are frequent omissions in the data as to whether clearances are cold or running, as to whether they apply to rotors or stators or both, and as to whether or not clearance/span was varied uniformly throughout multi-stage machines.

4. Data scatter is considerable.

5. In certain data, the reference performance level was not optimum.

6. In some data, maximum performance was shown for zero clearance, while in other data, maximum efficiency was achieved at relatively large clearance to span ratios.

Effect on Corrected Flow - Reduction of flow should probably be considered only for an inlet stage. No data was found for effects on this performance variable. The proposed model assumes that flow is reduced in direct proportion to one-half of the area difference between the tip clearance and the reference boundary layer displacement thickness, where the reference displacement thickness corresponds to the design assumption of annulus wall blockage. The derated flow is given by:

$$(Wa\sqrt{\theta/\delta})' = (Wa\sqrt{\theta/\delta}) \text{ ref } * \left[\frac{(a - d) * (1 - d)}{(a - c) * (1 - c)}\right]$$

for $b > c$, where

$Wa\sqrt{\theta/\delta}$ = corrected flow

$a$ = 2 Rm/h

$d$ = (b + c) / 2

Rm = mean average rotor passage radius, inches

b, c and h, as well as sub - and superscripts are as defined previously.

Effect on Pressure Ratio - The model assumes that stage pressure ratio is reduced as tip clearance increases beyond displacement thickness. The only data available was that of Reference 6. Pressure ratio decreases linearly according to the following relationship:

$$PR' = 1 + (PR_{ref} - 1) * (1 - 4.4\,(b - c))$$

for $b > c$, where

PR = stage pressure ratio

b, c, sub - and superscripts are as defined previously

#### 4.1.1.2 Effect on Blade Row Surface Finish for an Axial Stage

The data for surface finish effects on performance were obtained from References 2 and 6. The effect of increasing surface roughness is the degradation of all performance variables as the ratio of surface finish (rms in microinches) to chord (in inches) increases beyond a value of 42.7. Surface roughness is assumed to have no effect at lesser values. It is assumed that performance is reduced according to a linear function of this variable, based on empirical data.

Reference 6 showed that adverse effects of surface roughness decrease with increasing speed, while the opposite trend was observed in Reference 2. Reference 2 was used as the basis for the development to follow.

Effect on Efficiency - Peak adiabatic stage efficiency is estimated to be reduced according to

$$\eta ad' = \frac{\eta p,\ ref * (1.0316 - 0.00074 * (\mu/C)) - 9.064 * (PR - 1)}{1 - 8.887 * (PR - 1)}$$

for $\mu/C > 42.7$, where

$\mu$ = surface roughness, rms, microinches

C = blade chord, inches

Other variables are as previously defined.

Effect on Corrected Flow - The data shows corrected flow at peak efficiency to be reduced according to

$$(Wa\sqrt{\theta/\delta})' = (Wa\sqrt{\theta/\delta})\ ref * (1.021 - 0.000491 * (\mu/C))$$

for $\mu/C > 42.7$.

Effect on Pressure Ratio - Pressure ratio at peak efficiency is estimated to be reduced on the basis of

$$PR' = 1 + (PR_{ref} - 1) * (1.0316 - 0.00074* (\mu/C))$$

for $\mu/C$ 42.7.

#### 4.1.1.3 Effect of Rotor Solidity for an Axial Stage

Two data sources have been screened for the effect of solidity on stage performance.

References 7 and 8 document two NASA single-stage designs having rotor tip solidities of 1.3 and 1.7, respectively, for the same design point and annulus geometry. Design rotor pressure ratio was 1.800; adiabatic efficiency was 0.89; weight flow was 29.484 kg/sec; and tip speed was 422.888 m/sec. MCA blading was used. Rotor blade inlet metal angles were essentially the same for both designs. Transition point and exit blade metal angles differed, indicating the influence of solidity in the deviation rule used in the design. It is not known whether or not blade element loss and choke margin analyses differed between the two designs in relation to the different solidity levels. The following table summarizes the performance of the two rotors.

NASA 1.8 PR ROTOR PERFORMANCE

| | | |
|---|---|---|
| Tip Solidity | 1.3 | 1.7 |
| Peak Adiabatic Efficiency - % | 0.87 | 0.84 |
| *Corrected Flow, kg/sec | 29.5 | 28.5 |
| *Pressure Ratio | 1.80 | 1.79 |

*Values at peak efficiency

It can be seen that all performance variables were decreased at the higher solidity level. Beyond this generalization, however, specific correlation from this data is not deemed to be justifiable because of the potential influence of other design variables and the design system, i.e., choke and loss analyses.

The second data source was the overall performance of the three-stage Teledyne CAE NASA Research Compressor, which was tested at baseline, +30 percent and -10 percent solidities. Design speed performance is shown in Figure 4.1.1-2.

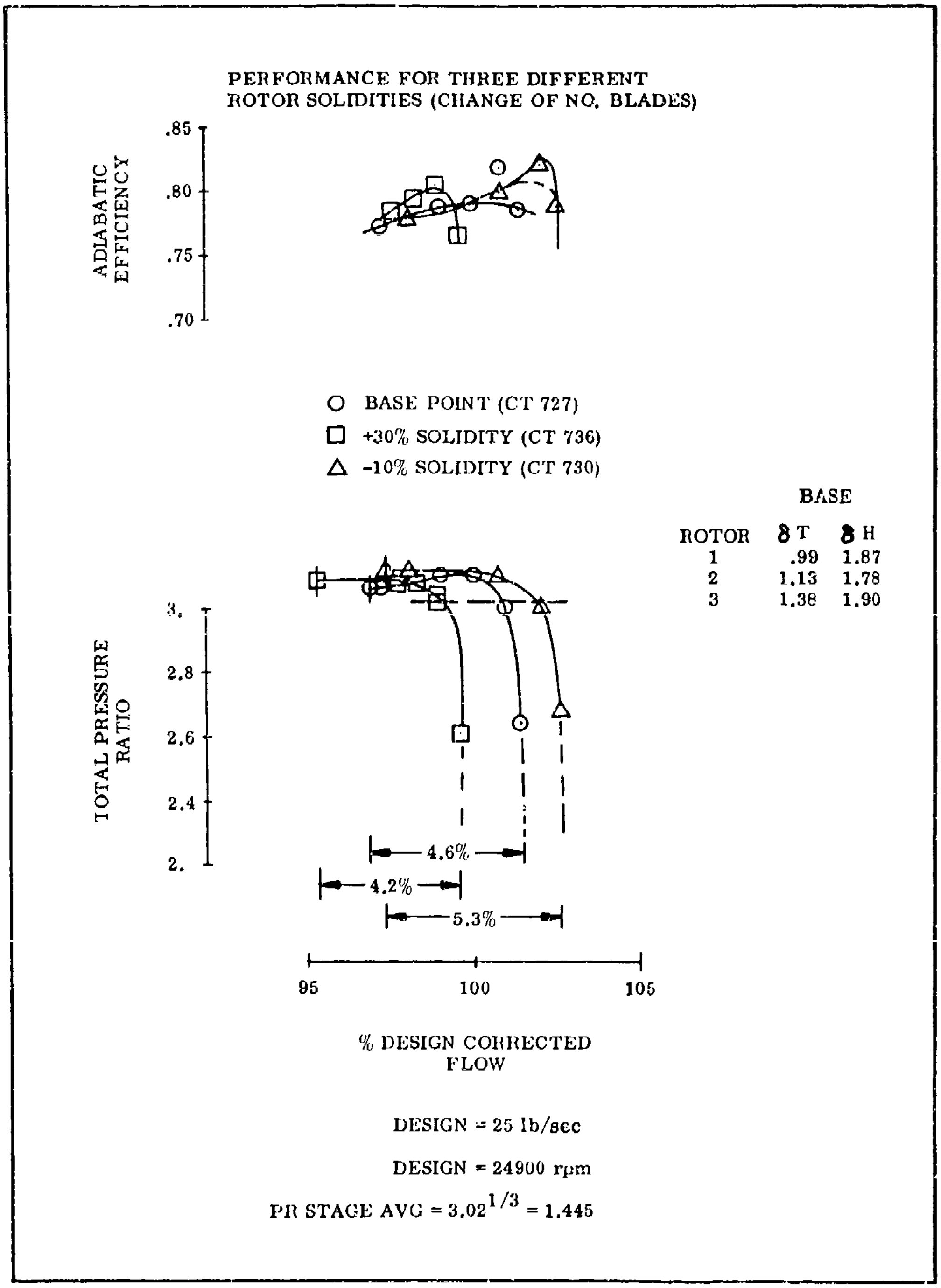


Figure 4.1.1-2. NASA Three-Stage Research Compressor.

In this case, solidity variation was obtained by varying the number of blades with no blade profile or angle changes. Maximum flow and efficiency occurred for the lowest solidity level and with the pressure ratio at peak efficiency remaining essentially constant. Flow range and surge margin were highest for the low solidity build. As in the previous case, specific correlation in terms of absolute solidity level is not considered to be justified, especially in view of the rotor-to-rotor solidity variation for this compressor and the very low value of rotor tip camber as compared to other engine rotor tip camber levels.

#### 4.1.1.4 Effect on Rotor Aspect Ratio for an Axial Stage

Reference 1 presents stage efficiency as a function of rotor clearance/span and aspect ratios for three different Reynolds numbers, as reproduced on Figure 4.1.1-3. Use of this data may be made for preliminary studies; however, such use should be tempered by a recognition that maximum stage average pressure ratio for the seven compressors surveyed in the reference was 1.18:1.

Further data is provided in Reference 9. Reproduced in Figure 4.1.1-4 is the design speed performance of two rotors having the same design point and with aspect ratios of 1.33 and 2.50. The wide-chord-rotor achieved design point flow and efficiency but had a low pressure ratio. Peak efficiency of the high aspect ratio rotor at 90 percent was about two points higher than for the low aspect ratio rotor. This difference is close to that indicated by Figure 4.1.1-3. Unfortunately, in this test, the tip solidity of the low aspect ratio rotor was 1.20 as opposed to 1.02 for the high aspect ratio rotor; the significance of this difference was not ascertained in the report. Beyond the scope of the present study, Figure 4.1.1-3 also illustrates the relative flow range extension typical of low aspect ratio rotors.

Teledyne CAE Turmo III-C aspect ratio test data was examined for consistency with the predictions of Figure 4.1.1-3. Because of questionable blade section definition, data for the 43-bladed rotor (aspect ratio of 2.47) was disregarded. However, between the 17- and 13-bladed rotors, with aspect ratios of 0.977 and 0.747, respectively, the design speed reduction of adiabatic efficiency at reduced aspect ratio was 2.5 percentage points (from 81.0 to 78.5), as compared to a predicted reduction of about 2.0 points fr m Figure 4.1.1-3. No significant difference in flow range was observed between the 17- and 13-bladed rotors.

#### 4.1.1.5 Effect of Blade Edge Thickness for Axial Stages

Reference 10 reports the results of an experimental investigation of the effect of rotor leading edge bluntness on the performance of the original stage as reported in Reference 7. Increased leading edge thickness was obtained by cutting back the blade leading edge such that the leading edge thickness was doubled at the tip. Axial extent of the cutback varied linearly to the hub. Tip solidity was reduced slightly.

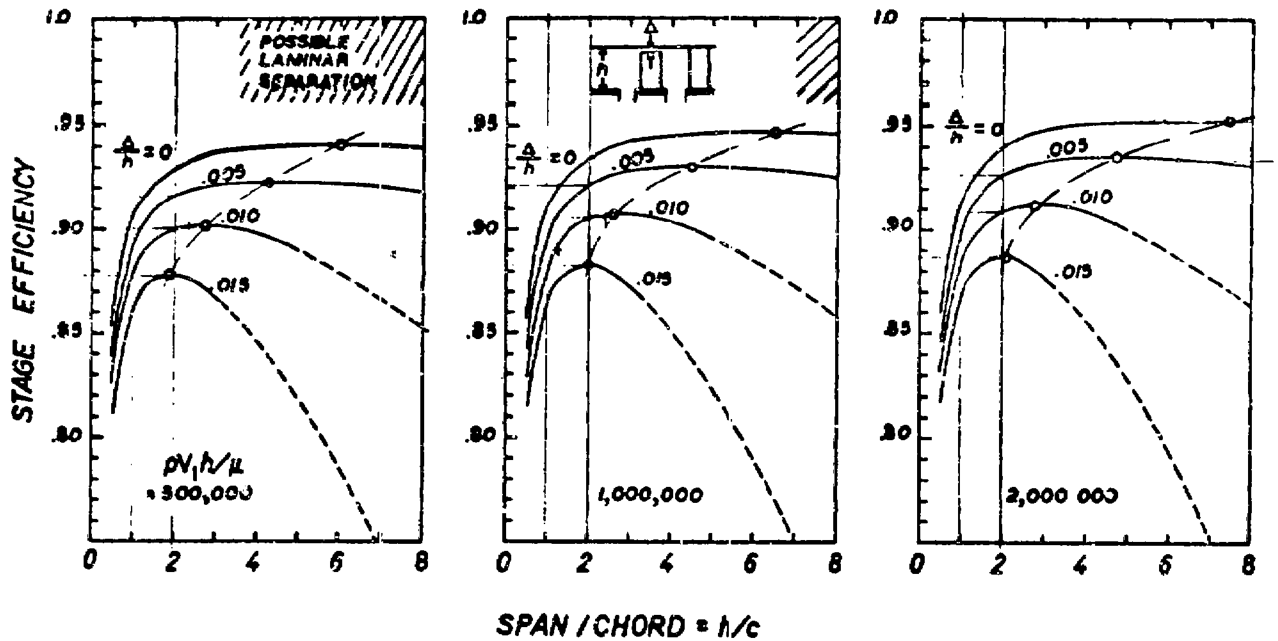


Figure 4.1.1-3. Stage Efficiency as a Function of Annulus Reynolds Number, Rotor Tip Clearance/Blade Span and Aspect Ratio.

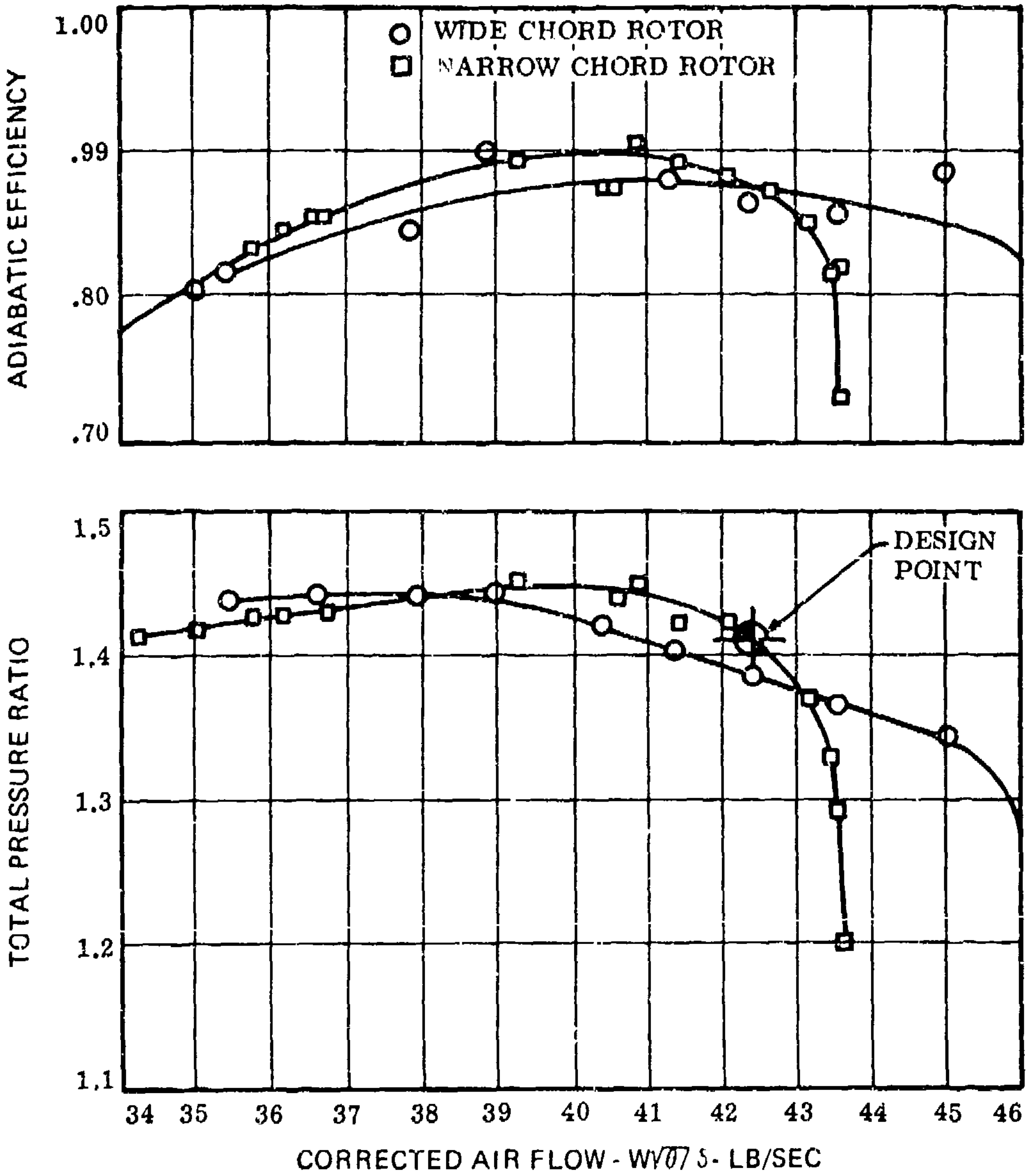


Figure 4.1.1-4. Comparison of Overall Performance Characteristics of Narrow-Chord and Wide-Chord Rotors at Design Tip Speed.

The increased leading edge thickness resulted in a decrease in rotor peak efficiency of about 3.5 points at design speed but had very little effect on efficiency at 90 and 70 percent design speed. Flow at design pressure ratio was reduced by about 3.0 lb/sec from 65.0 lb/sec. The principal conclusion of the report was that additional loss due to increased thickness resulting from the bow-wave system was small compared to the loss increase due to operation at a larger suction surface incidence. At design speed, the loss versus suction surface incidence followed the same general curve for both rotor configurations. By implication then, the report indicates that through proper incidence selection, loss increases can be limited to those related to leading edge blockage and the bow-wave system, as related in Reference 11.

The attainment of design flow rate for a stage (rotor) incorporating increased leading edge radii would require an upscale and/or blade profile redefinition. Attainment of design flow through the utilization of restaggered, open blading, would not be feasible because any flow increase would likely be accompanied by even higher incidence and loss levels.

Figure 4.1.1-5 shows a generalization of the effects of leading edge radius on rotor performance, based on design speed operation at peak adiabatic efficiency. The dashed lines indicated an estimate of achievable performance with upscale and/or blade redeisgn. Use of this data is not recommended unless absolutely necessary because correlation of rotor performance against blade tip geometry in this fashion is questionable. No similar generalization of the effects on rotor performance of leading edge bluntness were cited by the authors of Reference 1. The curves of Figure 4.1.1-5 are a linear fit of two points. If application of the data is a necessity, it should be done only in cases of rotors having like loading levels, tip inlet relative Mach numbers and hub/tip ratios.

The results of Reference 12, regarding rotor trailing edge thickness, showed no adverse effects due to increased blade trailing edge thickness to as much as 30 percent of maximum thickness for a subsonic compressor with 65 Series blading.

#### 4.1.1.6 Effect of Impeller Tip Clearance for a Centrifugal Stage

The effects of centrifugal compressor clearance control on efficiency improvement are shown in Figure 4.1.1-6.

Pressure ratio and efficiency variation with clearance, for constant surge margin, are shown on Figure 4.1.1-7 for a typical Teledyne CAE centrifugal stage at design corrected speed, based on the test data on Figure 4.1.1-8.

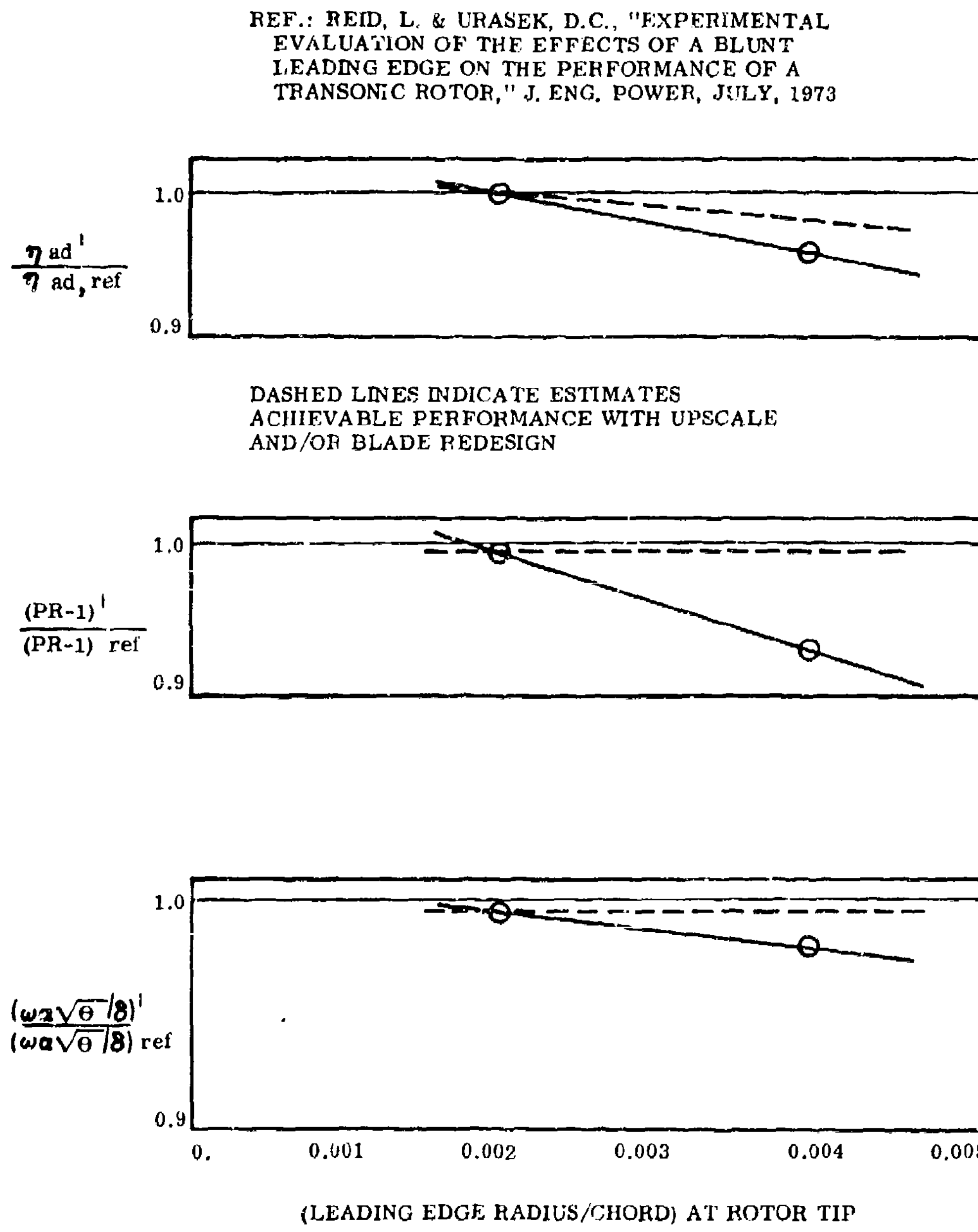


Figure 4.1.1-5. Effect of Leading Edge (Radius/Chord at Rotor Tip on Rotor Performance).

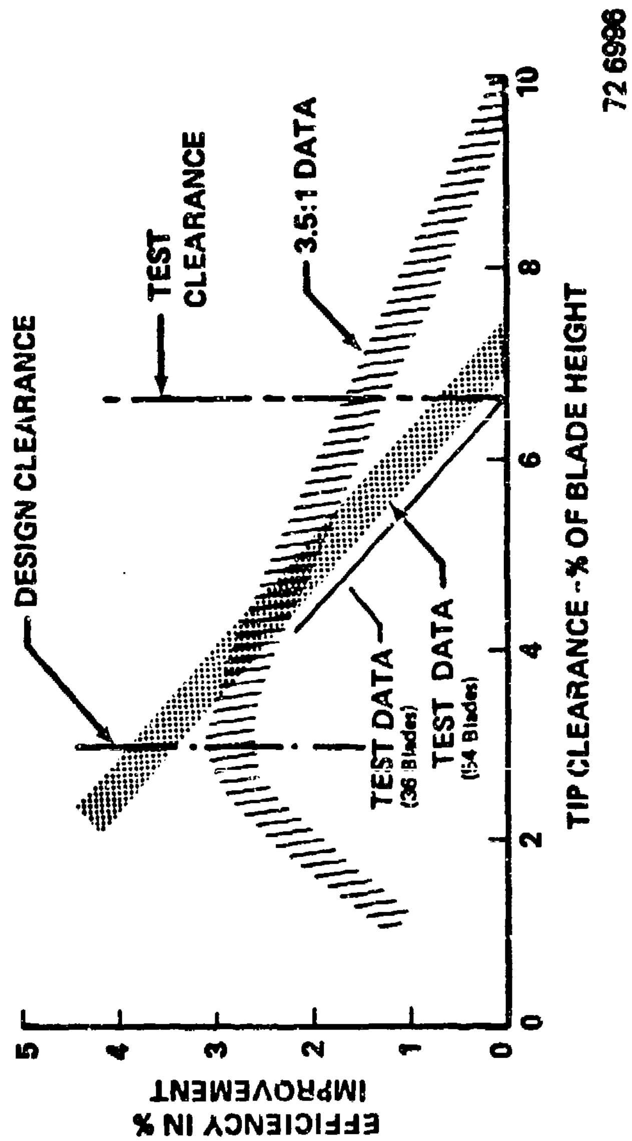


Figure 4.1.1-6. Centrifugal Compressor Clearance Control.

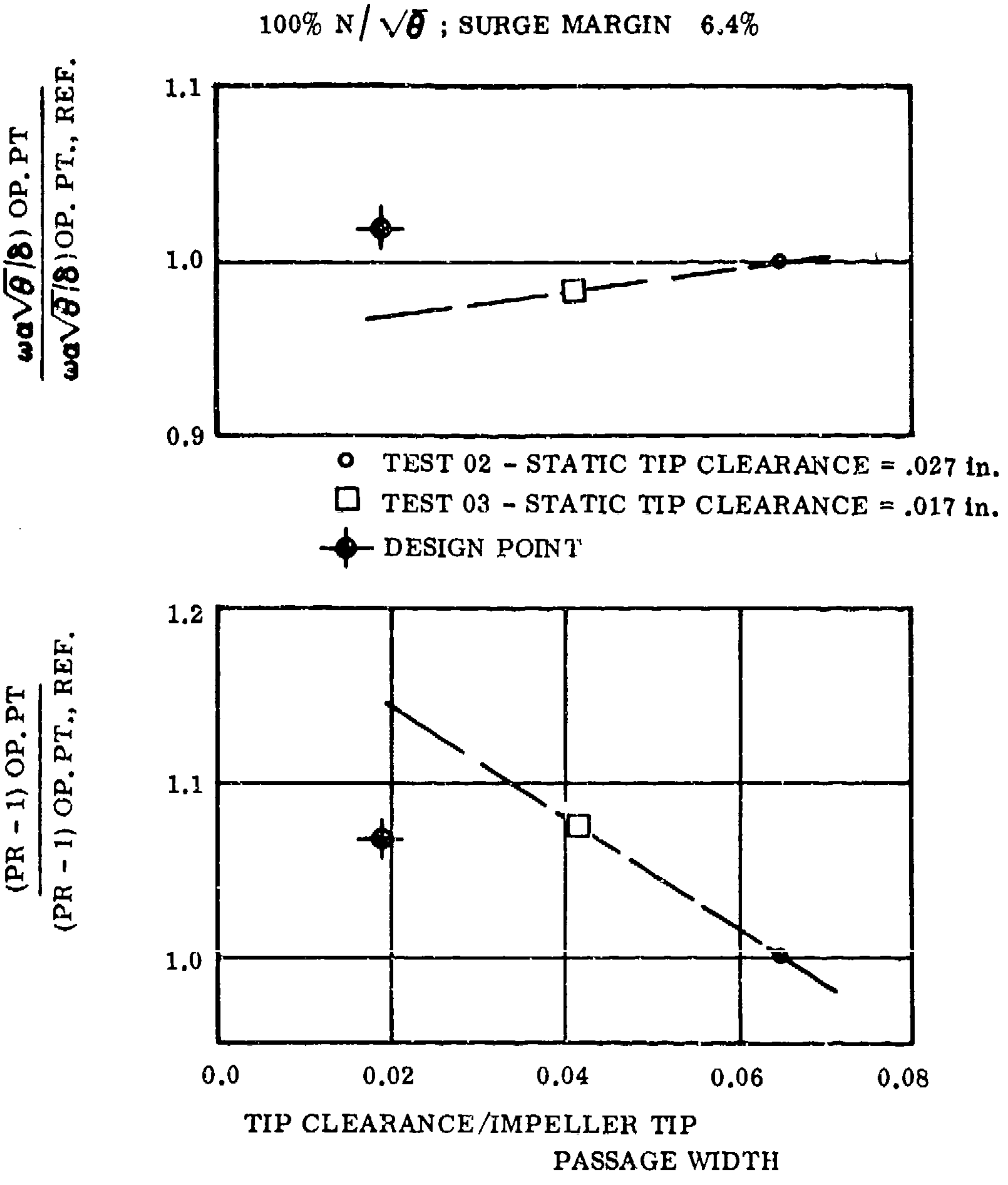


Figure 4.1.1-7. Effect of Impeller Tip Clearance/Impeller Tip Passage Width on Flow and Pressure Ratio.

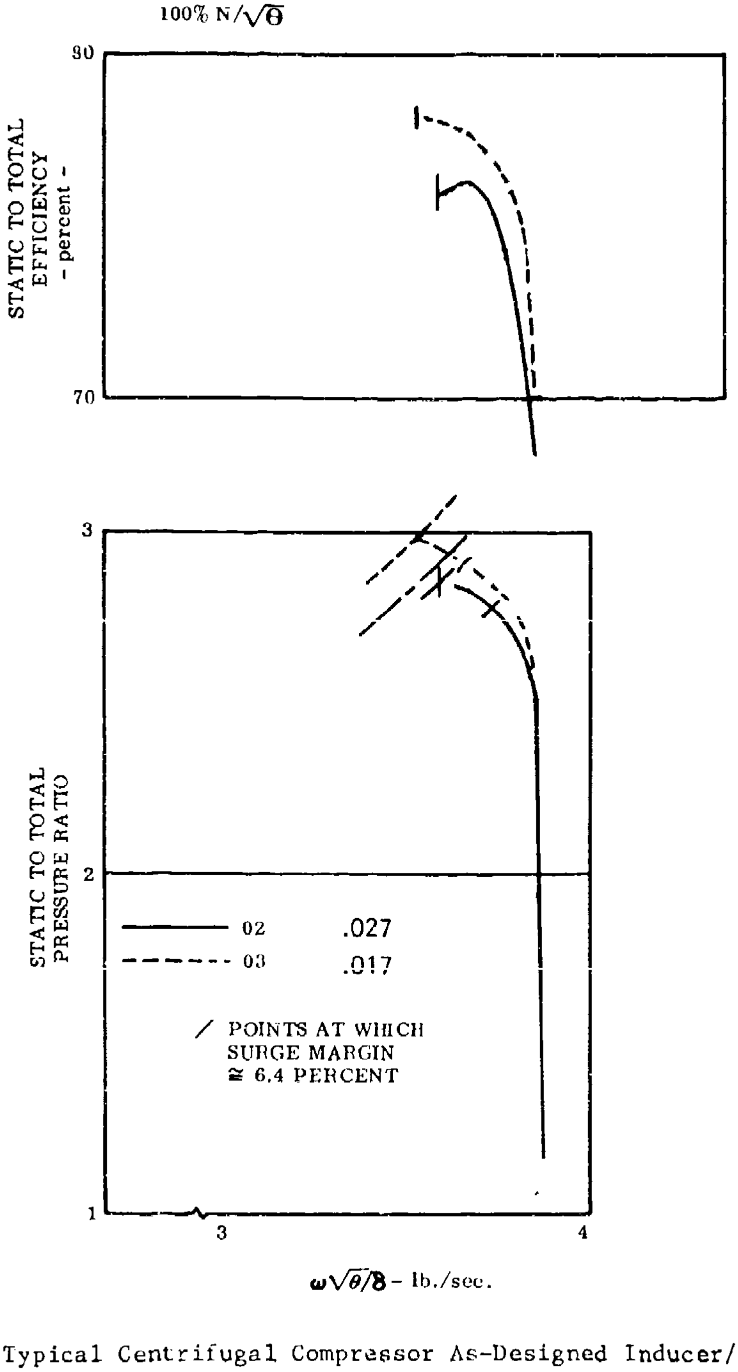


Figure 4.1.1-8. Typical Centrifugal Compressor As-Designed Inducer/Nominal Diffuser.

## 4.1.2 Reduced Cost Parameters for Turbines

An evaluation of cost-related turbine design parameters was conducted to determine their effect on turbine performance. The APSI LP turbine was used as a baseline for these analyses. A quantitative evaluation of performance effects can be related to cost through mechanical design and fabrication techniques using a DTC analysis approach.

### 4.1.2.1 Parametric Variation of Basic Geometric Variables

A systematic variation of the following turbine parameters was conducted:

1. Rotor exit swirl
2. Solidity of the nozzle and rotor
3. Rotor tip clearance
4. Rotor and nozzle trailing edge thickness
5. Rotor and nozzle aspect ratio
6. Counter-rotation of low pressure turbine with respect to the high pressure turbine
7. Combined effects along with rotor shrouding

The baseline LP turbine preliminary flowpath was sized to reflect an advanced turbine technology hub work coefficient goal. Rotor hub separation and high losses are avoided by maintaining positive reaction with high exit swirl. Investigations were conducted to optimize the performance with swirl. The design is premised on the use of an exit guide vane downstream of the turbine to recover the rotational energy from the rotor.

Figures 4.1.2-1 through 4.1.2-4 demonstrate that efficiency can be increased by appropriately reducing the rotor tip clearance, nozzle and rotor trailing edge thickness, and the number of vanes and blades. Aspect ratio effects are shown in Figures 4.1.2-5 through 4.1.2-8.

Figure 4.1.2-9 demonstrates the combined effects of reducing previously defined parameters. At 40 degrees exit swirl, the efficiency is marginal; and with the expected degradation of the exit guide vane, the stage-and-a-half efficiency would be less.

To further investigate increases in efficiency of the low pressure turbine, counter-rotation and the use of a rotor shroud were investigated. The results of this investigation are shown in Figure 4.1.2-9 by the point at the 40 degree exit swirl. Two points have been gained.

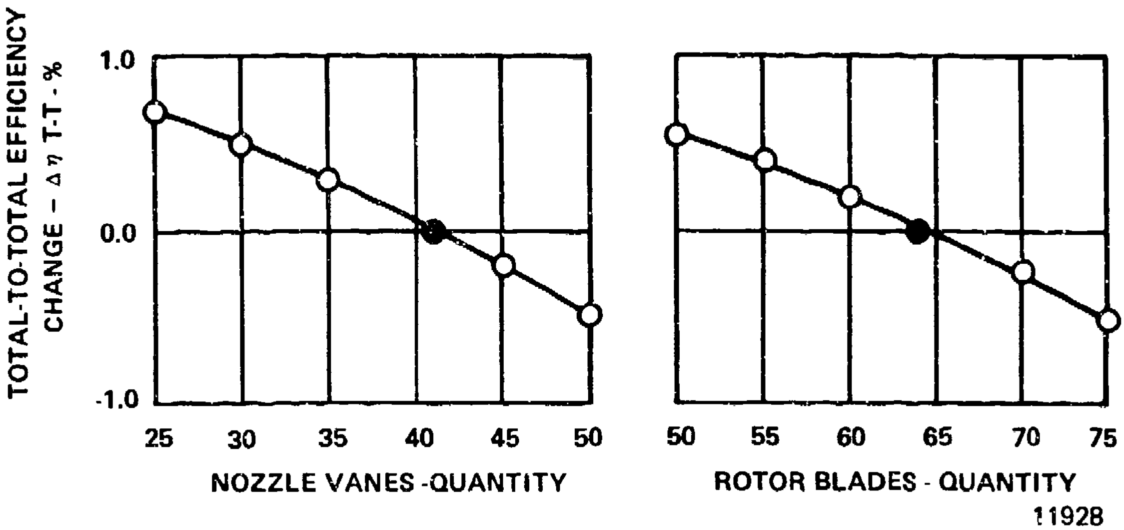


Figure 4.1.2-1. Effect of Number of Vanes and Blades on Stage Efficiency.

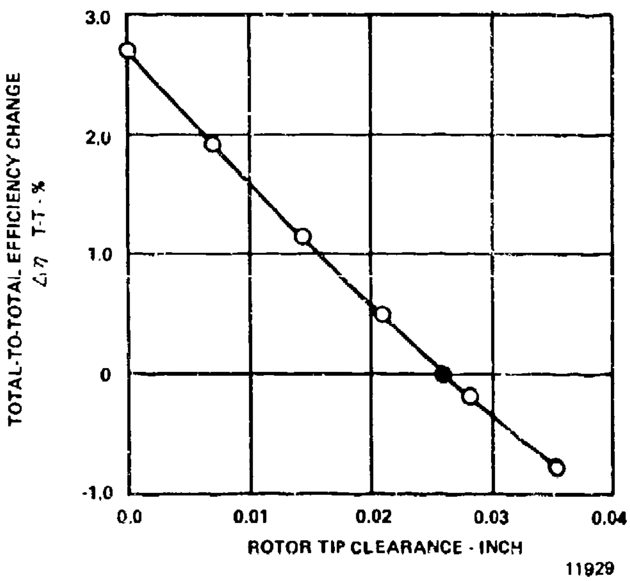


Figure 4.1.2-2. Effect of Rotor Tip Clearance on Stage Efficiency.

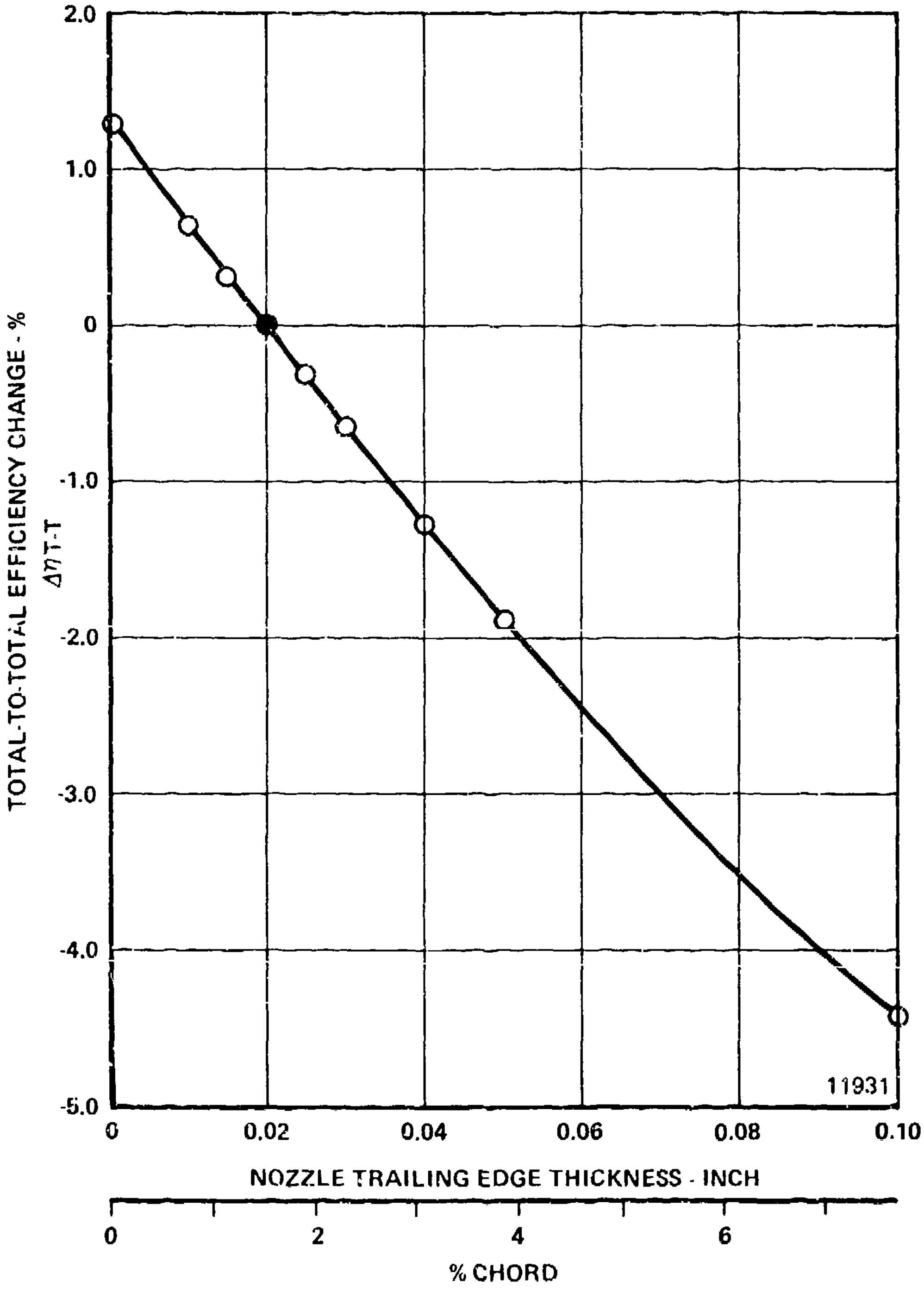


Figure 4.1.2-3. Effect of Nozzle Trailing Edge Thickness on Stage Efficiency.

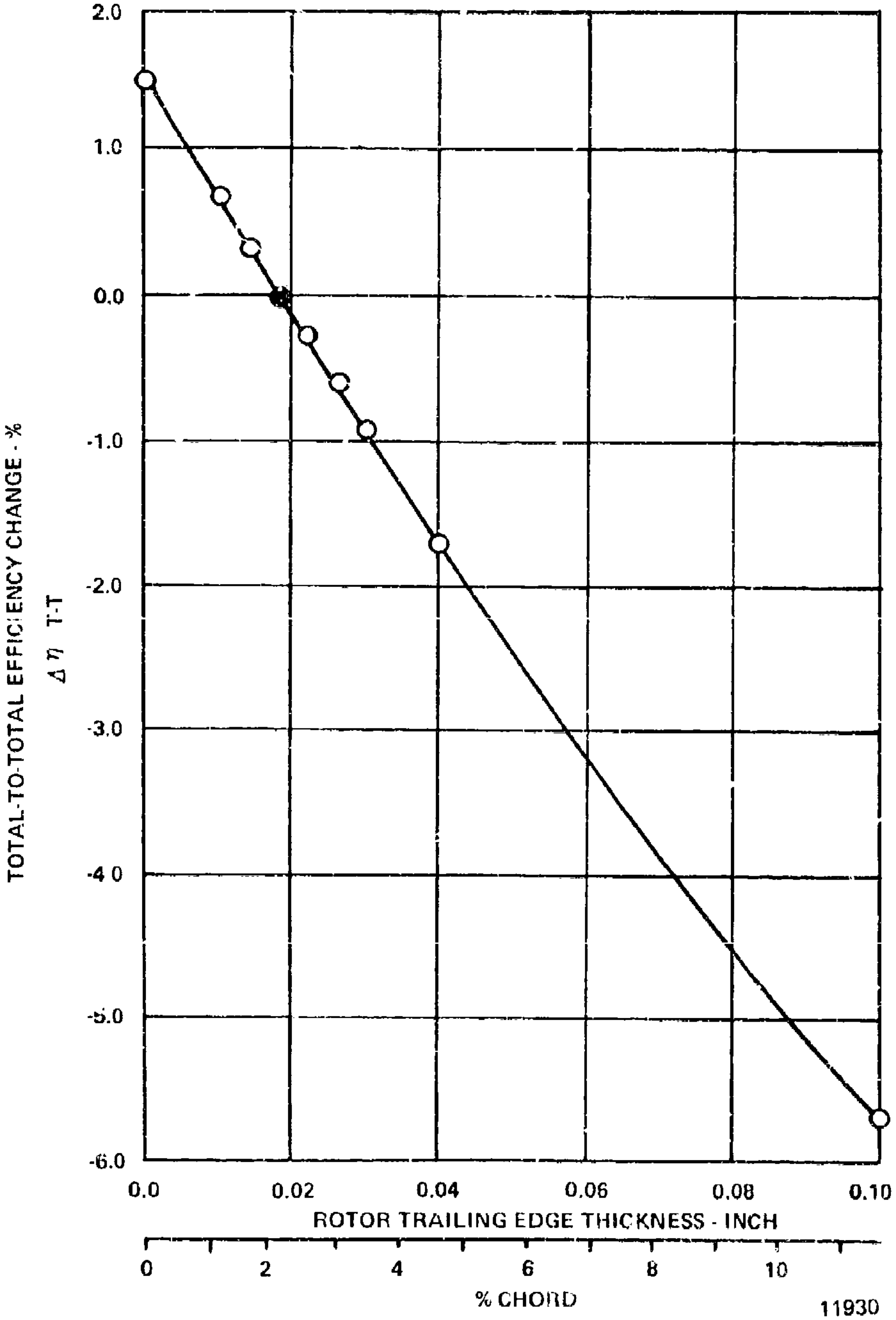


Figure 4.1.2-4. Effect of Rotor Trailing Edge Thickness on Stage Efficiency.

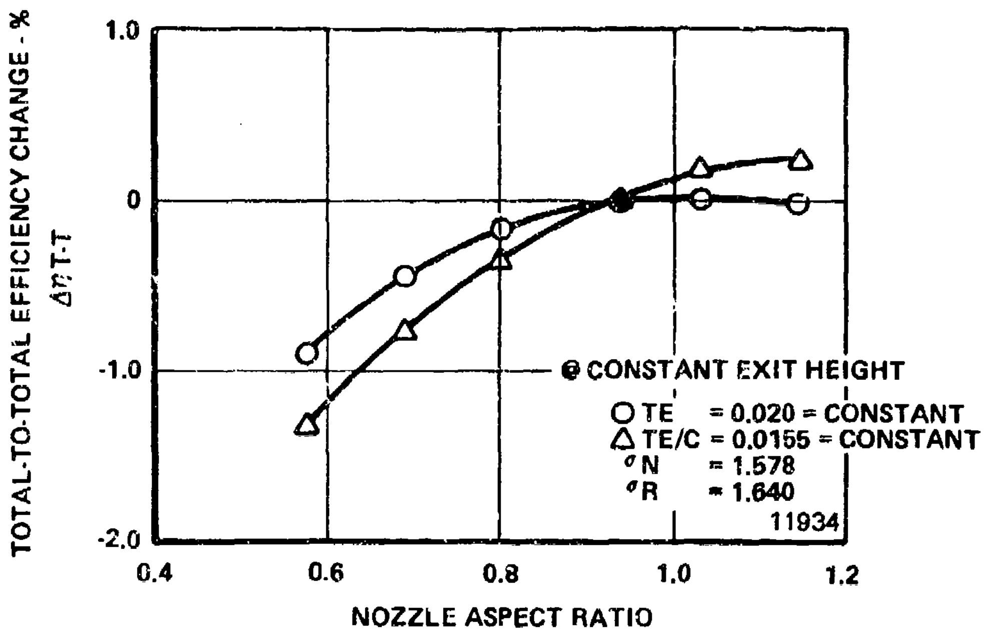


Figure 4.1.2-5. Effect of Stage Efficiency by Varying Nozzle Aspect Ratio by Chord n- While Maintaining Constant Solidity and Maintaining Similar Blade Surface Velocity Distribution.

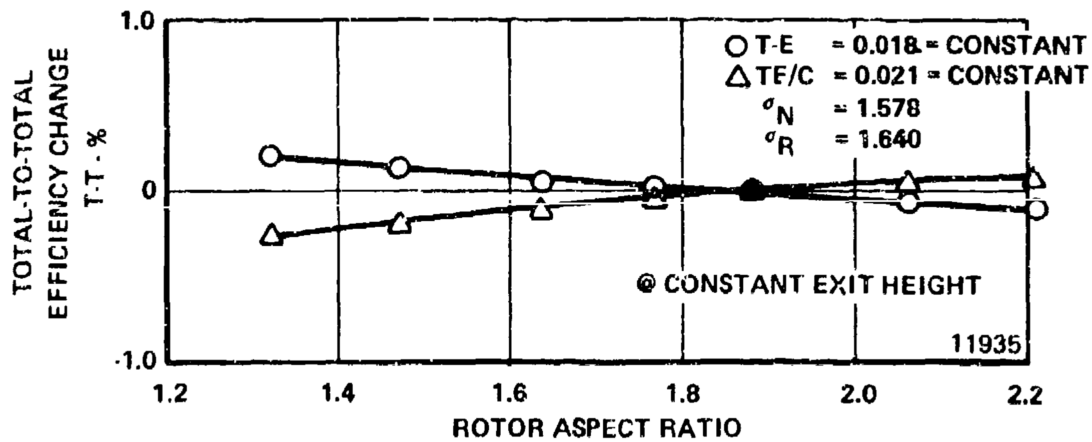


Figure 4.1.2-6. Effect of Stage Efficiency by Varying Rotor Aspect Ratio by Chord n- While Maintaining Constant Solidity and Maintaining Similar Blade Surface Velocity Distribution.

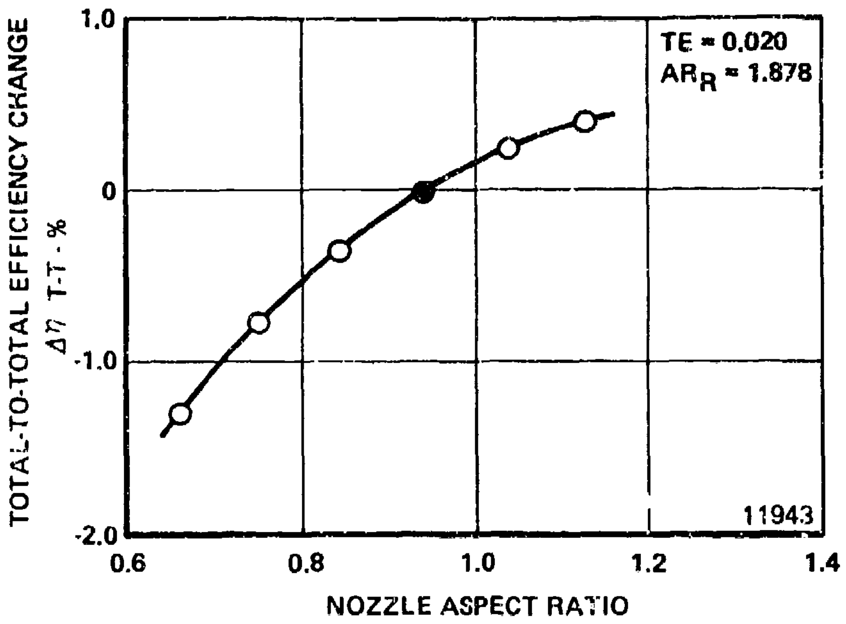


Figure 4.1.2-7. Effect on Efficiency by Varying Nozzle Aspect Ratio by Vane Height.

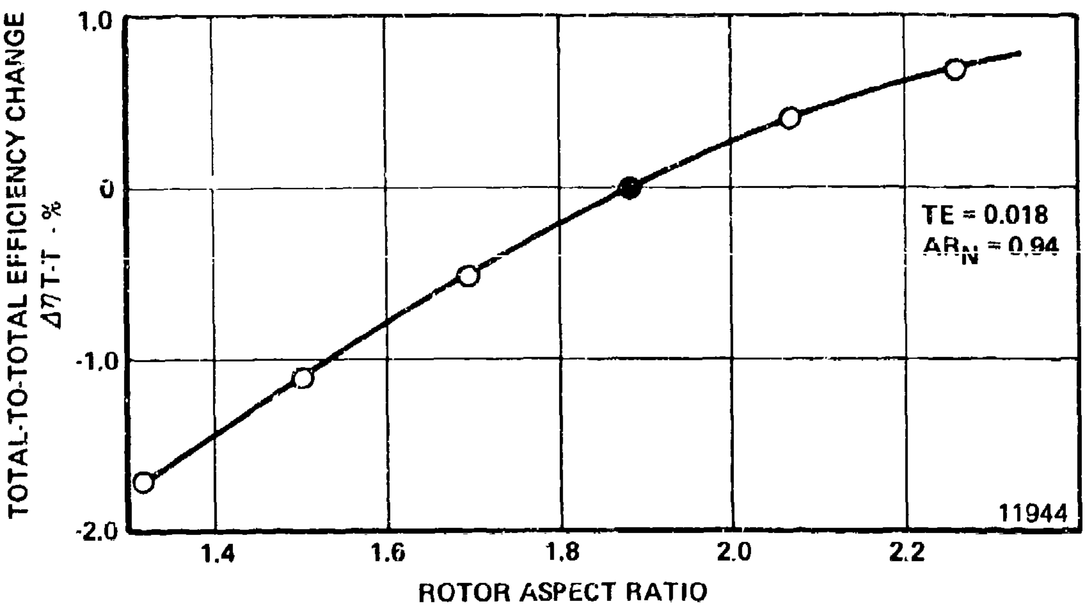


Figure 4.1.2-8. Effect on Efficiency by Varying Rotor Aspect Ratio by Blade Height.

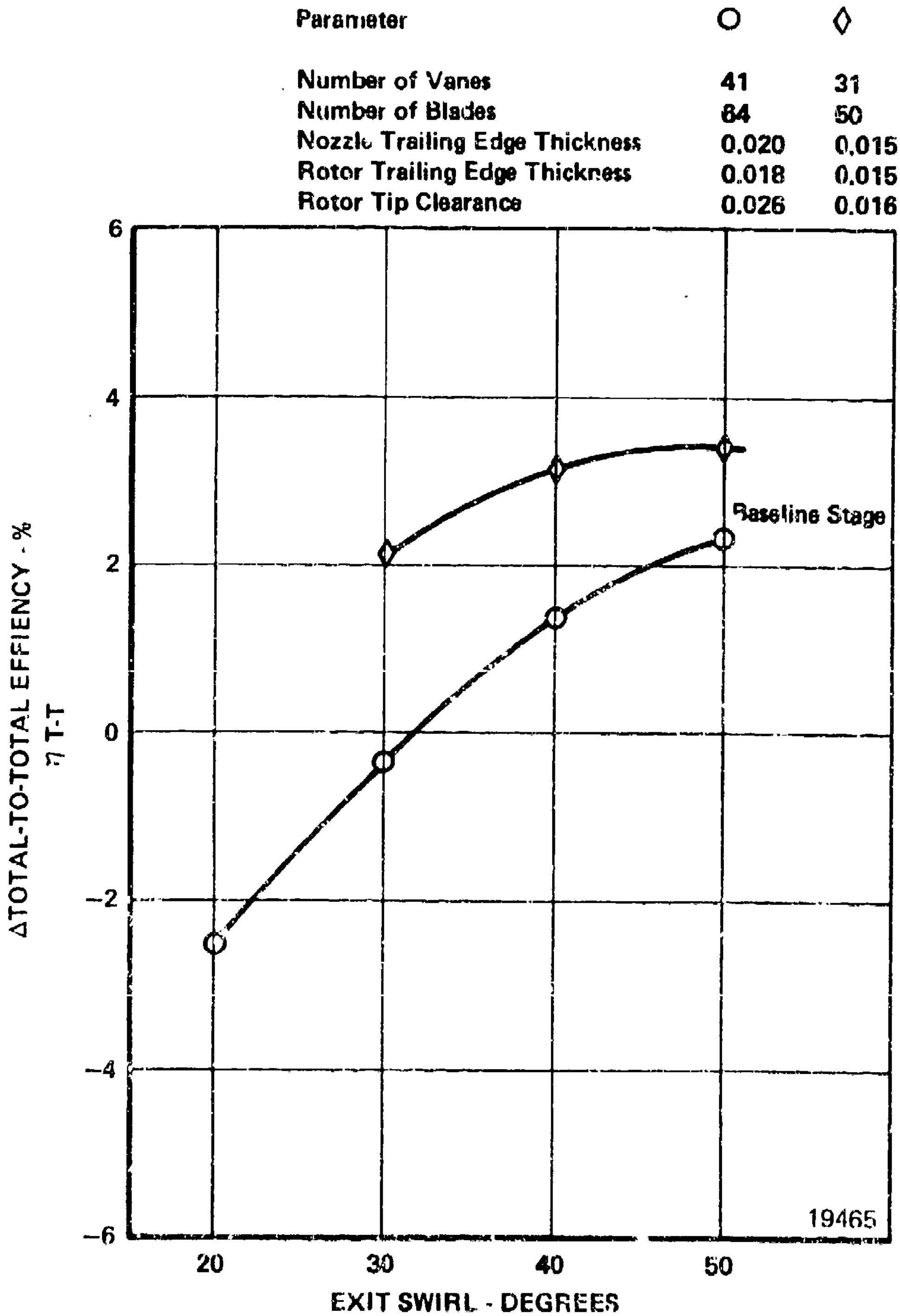


Figure 4.1.2-9. Combined Effects of Hardware Changes on Efficiency of LP Turbine.

## 4.2 Cost Reduction Topics

The candidate cost reduction topics were obtained by a survey of all specialized engineering and manufacturing disciplines within Teledyne CAE (including compressor design, turbine design, structural analysis, controls and accessories, test engineering, manufacturing methods, and materials technology) for ideas offering cost reduction and/or performance improvement. The topics thus gathered were screened by selection process based on potential benefits to be derived in relation to the probability of successful implementation. The topics selected for further investigation were:

Fan Blade Design - Effect of blade aspect ratio on total cost, weight and performance for the same blade solidity factor.

Fan Blade Materials - Cost and weight comparison of forged and machined titanium blades versus forged and machined steel blades.

Cast Titanium Components - Applicability of cast titanium to various engine components including fan blades, compressor hubs, etc., to reduce engine cost and weight.

Be-Ti Fan Power Shaft - Engine weight reduction as a result of a lighter shaft and the smaller shaft diameter permitted by the lower density-higher modulus beryllium/titanium powdered metal alloy.

Brazed Axial Compressor - Brazed assembly using titanium sheet metal blades to reduce compressor fabrication costs.

Hybrid Radial Compressor Diffuser - Combining both the radial and axial sections of a standard diffuser into a single cascade element; offers cost, weight, and engine envelope advantages.

Infusion Cooled Vaporizer Plate Combustor - Lowering the costs of fabrication by using fewer fuel injection points through more efficient fuel spreading and vaporization.

Ceramic Components - Investigation of cost reductions to be obtained from uncooled ceramic components formed to size as compared to conventional cooled hardware for combustors and turbines.

Powdered Metal Components - Applicability of powdered metal technology to various engine components, particularly formed-to-finish-size turbine blades and discs.

Eliminate LP Turbine Inlet Stator - Feasibility of running contra-rotating shafts without an inlet stator to the LP turbine stages.

Welded LP Turbine Assembly - Cost advantages of joining two (or more) turbine stages by EB welding.

Jet-Flap Turbine Blading - Feasibility of extending the jet-flap concept to the fan turbine blading.

Inter-Shaft Sealing - Elimination of the two inboard oil seals in the front bearing cavity by providing an inter-shaft gas seal.

Gas Foil Bearings - The feasibility of replacing rolling element bearings and the accompanying lubrication requirements to provide cost advantages.

Simplified Lube System - Cost reductions by eliminating multiple element scavenge pumps and using simple, high-speed rotating pumping elements within each bearing cavity.

High Speed Accessory Drive - Cost, weight and performance improvements by using a reduced size drive shaft and strut through the inlet flow path.

Controls and Accessories - Reduction in cost and weight of three control components.

Engine Specification Revisions - Specification changes to reduce engine cost and weight for specific engine applications.

### 4.2.1 Fan Blade Design

The objective of this study was to determine the effect of aspect ratio (blade height to mean chord) on the total cost and weight of a typical titanium fan stage. The study analyzed designs using blade aspect ratios of 1.0, 2.0, 3.0 and 4.0; each design had the same blade solidity factor. In addition, the analysis was repeated for fan stages geometrically scaled by 0.707 (half-thrust size) and 1.414 (twice-thrust size). Based on the constant solidity factor, the number of blades corresponding to each aspect ratio were determined to be:

Aspect Ratio of 1.0 - 14 blades
Aspect Ratio of 2.0 - 28 blades
Aspect Ratio of 3.0 - 42 blades
Aspect Ratio of 4.0 - 56 blades

A generalized method for estimating the machining and material cost of a titanium blade was developed. The cost factors assumed are the following:

| | |
|---|---|
| Finishing of the airfoil surface | $ 15.75 per 10 square inches |
| Dovetail grinding | $ 3.00 per blade |
| Polishing | $ 1.25 per inch of blade height |
| Part shroud finishing, coating, heat treat, etc. | $ 15.00 per blade |
| Forging material | $ 8.40 per pound |

The blade and dovetail geometry and support discs were sized and drawings were made of all twelve configurations (four aspect ratios, three scale sizes). Using these drawings, the finished weight, forging weight, and airfoil surface area were calculated for each blade. The results of these calculations are:

| | Aspect Ratio | Finished Weight | Forging Weight | Airfoil Surface Area ($in^2$) |
|---|---|---|---|---|
| Half-Thrust Size | 1.00 | 1.01 | 1.51 | 44.6 |
| | 2.00 | 0.25 | 0.37 | 22.3 |
| | 3.00 | 0.11 | 0.17 | 14.9 |
| | 4.00 | 0.06 | 0.09 | 11.1 |
| Basic Thrust Size | 1.00 | 2.85 | 4.28 | 88.9 |
| | 2.00 | 0.70 | 1.06 | 44.4 |
| | 3.00 | 0.32 | 0.47 | 29.7 |
| | 4.00 | 0.17 | 0.26 | 22.2 |
| Twice-Thrust Size | 1.00 | 8.07 | 12.10 | 178.0 |
| | 2.00 | 2.00 | 3.00 | 89.0 |
| | 3.00 | 0.89 | 1.34 | 59.4 |
| | 4.00 | 0.50 | 0.75 | 45.0 |

In addition, a generalized method for estimating the machining and material cost for the support disc was developed. The cost factors assumed were the following:

| | |
|---|---|
| Single point turning material removal | $ 7.70 per 100 cubic inches |
| Dovetail broaching | $ 1.50 per dovetail slot |
| Forging material | $ 6.60 per pound |

The cost for the various configurations of the fan blades and support disc for each of the twelve cases considered were calculated using the generalized methods described. The forging weights, finished weights, machining costs and material costs for each case are summarized in Figure 4.2.1-1.

The fan rotor weight with titanium blades versus aspect ratio has been plotted as shown on Figure 4.2.1-2. As expected, the fan rotor weight decreases with an increase in aspect ratio. The plot also shows, particularly for the lower thrust engines, that further increases in aspect ratio effect relatively minor changes in the weight of the fan rotor. The fan rotor cost versus aspect ratio plotted in Figure 4.2.1-3 shows that there is an optimum aspect ratio for each thrust size to provide a minimum actual rotor cost.

The impact of only the weight differential showed it is advisable to use the highest aspect ratio possible. Further investigation showed that there is performance degradation which must be addressed to allow for fabrication of the higher aspect ratio blades. A higher aspect ratio blade has a shorter chord and thinner blade and requires smaller leading edge and trailing edge radii. Fabrication of the blades with smaller radii reaches a practical limit when a minimum radius is reached. Higher aspect ratio blades can only be fabricated in small sizes by compromising leading edge radius. These blades therefore suffer a performance degradation which equates to an increase in specific fuel consumption, a detrimental system cost impact. The impact of both weight and SFC have been plotted for the different aspect ratios as they relate to the baseline engine (Figure 4.2.1-4), the half-thrust engine (Figure 4.2.1-5) and the twice-thrust engine (Figure 4.2.1-6). Additional rotor costs are also affected by the use of a part-span shroud which is required to maintain the structural integrity of the rotor at aspect ratios normally between 1.75 and 2.5. This is plotted in the above referenced curves because the actual aspect ratio where the shroud is required must be determined by the specific blade configuration. These curves illustrate the following points:

Small engines require lower aspect ratios. (Performance degradation occurs sooner and to a greater extent.)

Conversely, larger engines should use higher aspect ratios.

For any size engine, there is an optimum aspect ratio to provide lowest system cost.

These curves can be used to identify the blade aspect ratio that provides the lowest system cost; however, other considerations, such as structural integrity, may be the determining factor in aspect ratio selection.

| Scale Factor | Aspect Ratio | No. of Blades | Compressor Blades (Titanium): Material Weight | Compressor Blades (Titanium): Material Cost | Blade Machining: Dovetail Grinding | Blade Machining: Airfoil Grinding | Blade Machining: Leading and Trailing Edge | Blade Machining: Part Span Shroud | Compressor Blades (Titanium): Finish Weight | Compressor Blades (Titanium): Cost/Blade (Less Part Span Shroud) | Compressor Hub (Titanium): Material Weight | Compressor Hub (Titanium): Material Cost | Hub Machining: Mach | Hub Machining: Broach | Compressor Hub (Titanium): Finish Weight | Compressor Hub (Titanium): Cost/Hub | Compressor Rotor Assembly: Total Weight | Compressor Rotor Assembly: Total Cost Less Part Span Shroud | Compressor Rotor Assembly: Total Cost With Part Span Shroud |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| 0.5 | 1 | 14 | 1.51 | $12.68 | $3.00 | $69.40 | $5.25 | $15.00 | 1.01 | 90.33 | 40.88 | $228.90 | $16.30 | $21.00 | 7.45 | $266.20 | 21.58 | $1530.82 | $1740.82 |
| | 2 | 28 | 0.375 | 3.15 | | 34.50 | | | 0.25 | 45.90 | 25.86 | 144.80 | 10.30 | 42.00 | 4.58 | 197.10 | 11.58 | 1482.30 | 1902.30 |
| | 3 | 42 | 0.168 | 1.41 | | 23.07 | | | 0.112 | 32.73 | 20.52 | 114.90 | 8.40 | 63.00 | 3.02 | 186.30 | 7.71 | 1560.96 | 2190.96 |
| | 4 | 56 | 0.095 | 0.79 | | 17.34 | | | 0.063 | 26.38 | 17.21 | 96.40 | 7.07 | 84.00 | 2.67 | 187.47 | 6.20 | 1664.75 | 2504.75 |
| 1.0 | 1 | 14 | 4.28 | 35.99 | 3.00 | 137.78 | 7.50 | 15.00 | 2.86 | 184.27 | 93.04 | 549.00 | 34.00 | 21.00 | 28.16 | 604.00 | 68.15 | 3183.78 | 3393.78 |
| | 2 | 28 | 1.06 | 8.91 | | 69.80 | | | 0.707 | 89.21 | 56.12 | 308.70 | 21.00 | 42.00 | 12.58 | 371.70 | 32.38 | 2869.58 | 3289.58 |
| | 3 | 42 | 0.474 | 4.00 | | 45.10 | | | 0.316 | 60.60 | 43.27 | 242.30 | 15.50 | 63.00 | 8.28 | 320.80 | 21.54 | 2866.00 | 3498.00 |
| | 4 | 56 | 0.268 | 2.25 | | 34.72 | | | 0.179 | 47.47 | 36.49 | 204.30 | 14.10 | 84.00 | 7.40 | 302.40 | 17.41 | 2960.72 | 3800.72 |
| 2.0 | 1 | 14 | 12.1 | 101.74 | 3.00 | 277.53 | 10.50 | 15.00 | 8.07 | 392.77 | 205.03 | 1143.20 | 65.00 | 21.00 | 71.31 | 1234.20 | 184.35 | 6732.98 | 6942.98 |
| | 2 | 28 | 3.00 | 25.20 | | 138.17 | | | 2.00 | 176.87 | 127.68 | 715.00 | 44.60 | 42.00 | 35.90 | 801.60 | 91.90 | 5753.96 | 6173.96 |
| | 3 | 42 | 1.34 | 11.26 | | 92.37 | | | 0.894 | 117.13 | 93.65 | 524.40 | 33.90 | 63.00 | 23.63 | 621.30 | 61.19 | 5540.76 | 6170.76 |
| | 4 | 56 | 0.758 | 6.35 | | 69.38 | | | 0.505 | 89.24 | 78.45 | 439.50 | 28.30 | 84.00 | 20.32 | 551.80 | 48.60 | 5549.24 | 6389.24 |

Figure 4.2.1-1. Titanium Fan Blade Cost Analysis Summary.

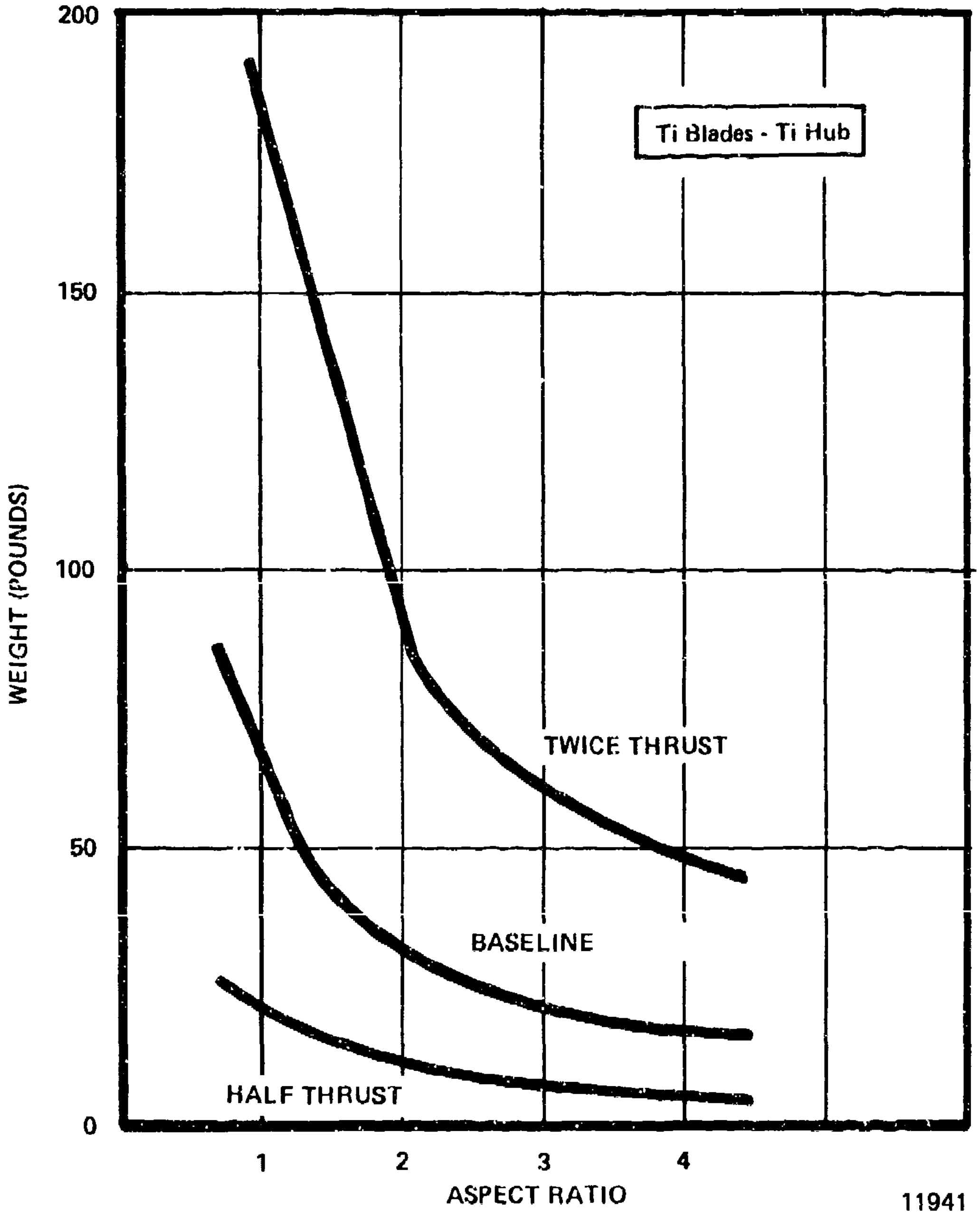


Figure 4.2.1-2. Fan Rotor Weight Versus Aspect Ratio - Titanium Blades and Hub.

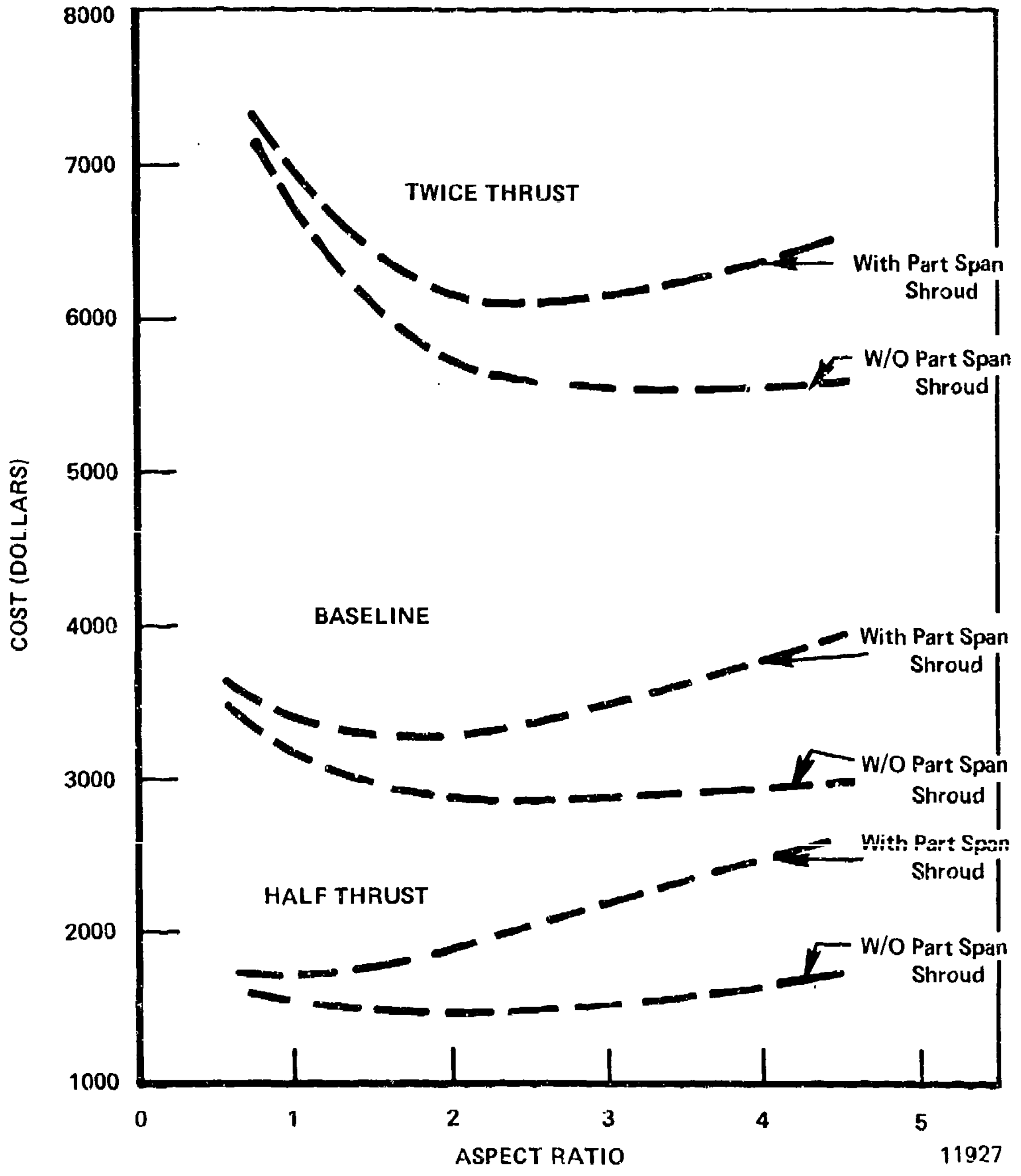


Figure 4.2.1-3. Fan Rotor Cost Versus Aspect Ratio - Titanium Blades and Hub.

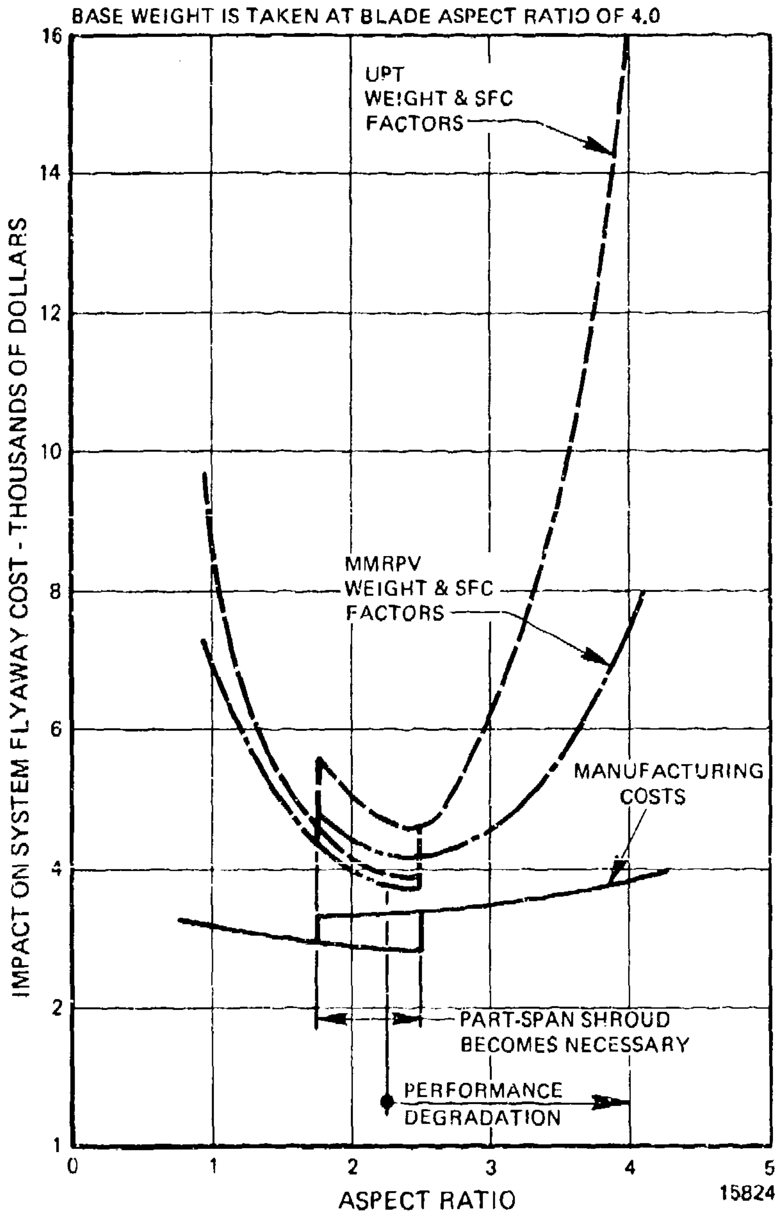


Figure 4.2.1-4. Fan Rotor Cost Versus Blade Aspect Ratio - Baseline Thrust Engine, Ti Blades/Ti Hub.

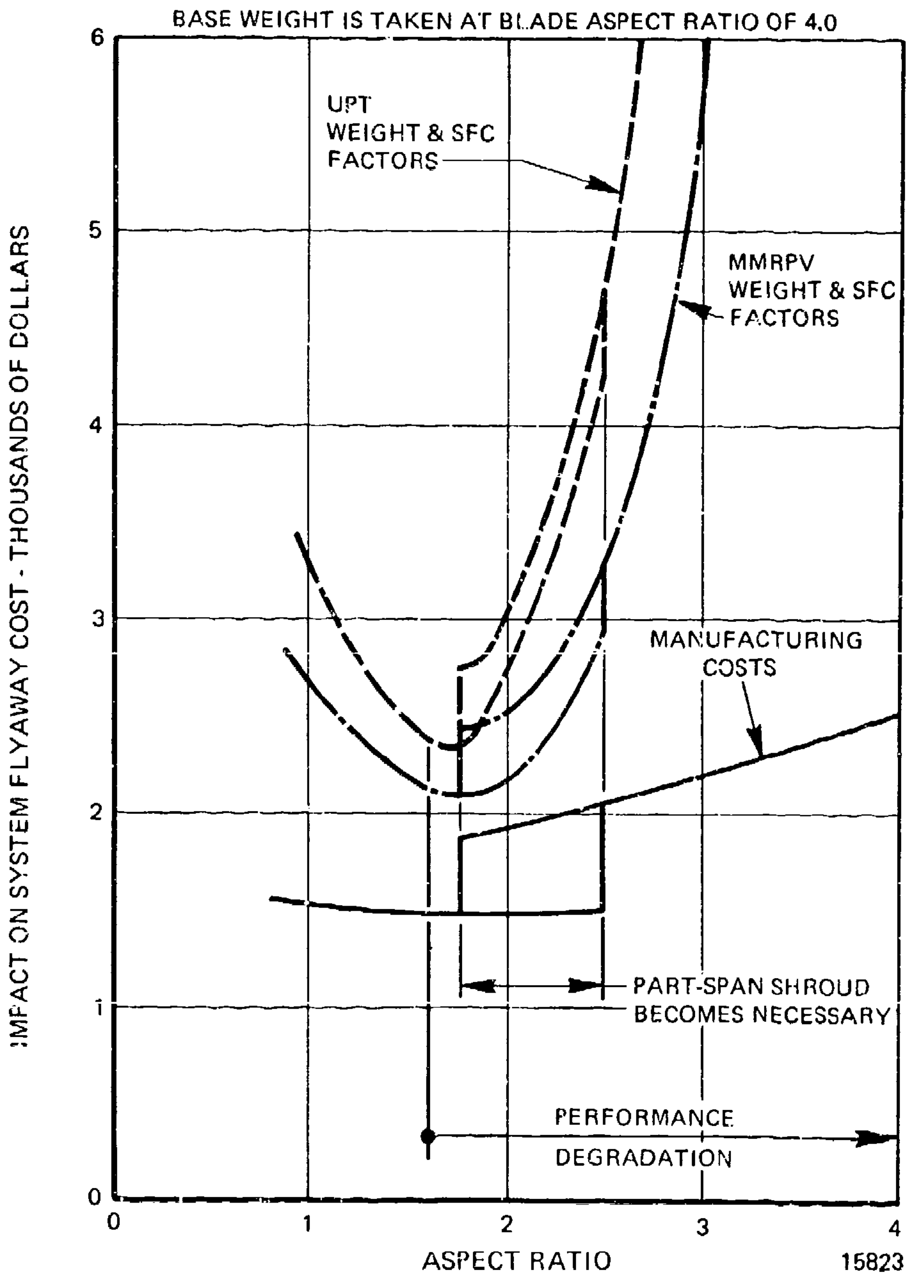


Figure 4.2.1-5. Fan Rotor Cost Versus Blade Aspect Ratio - Half-Thrust Engine, Ti Blades/Ti Hub.

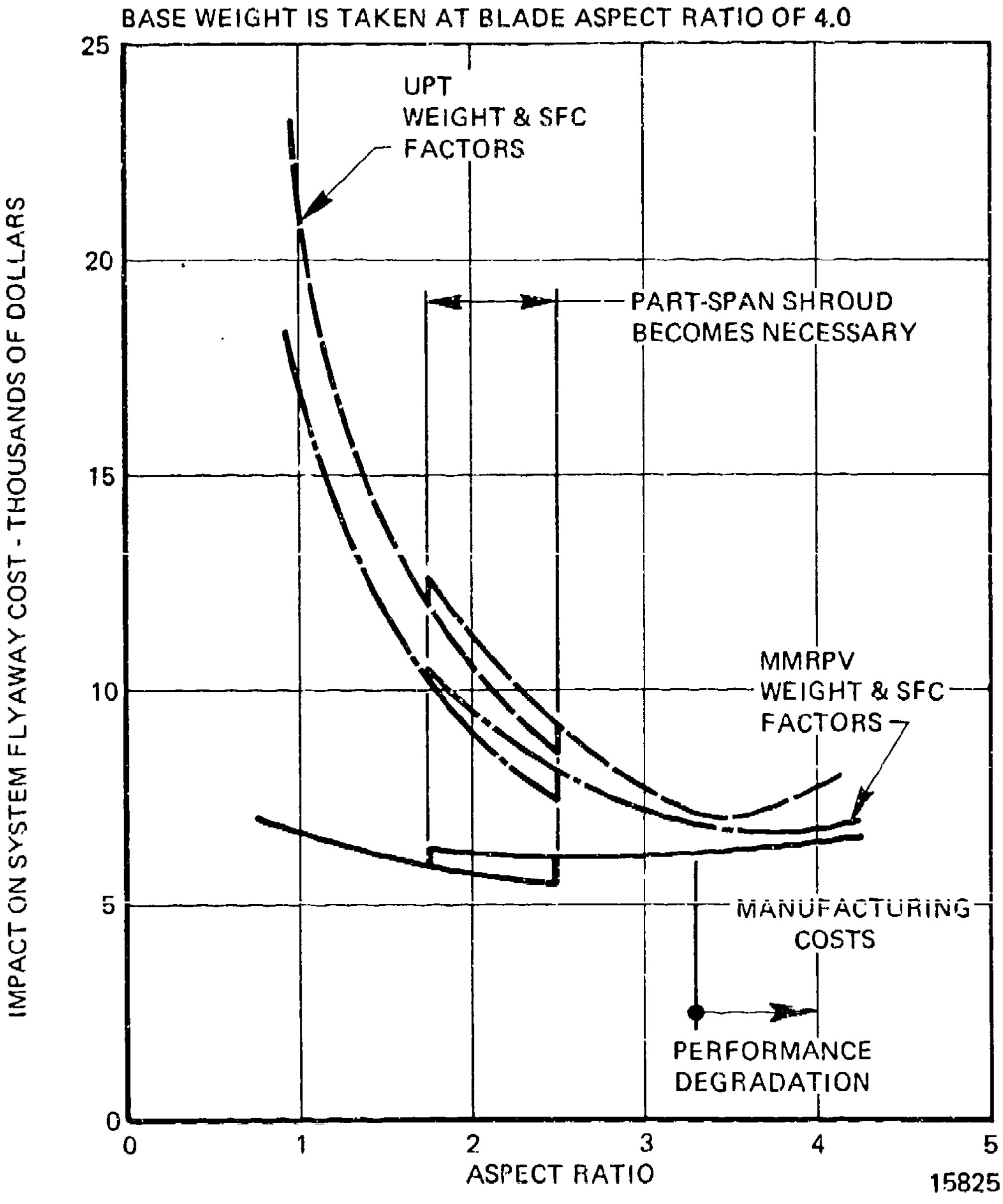


Figure 4.2.1-6. Ian Rotor Cost Versus Blade Aspect Ratio - Twice-Thrust Engine, Ti Blades/Ti Hub.

### 4.2.2 Fan Blade Materials

The objective of this study was to determine the cost and weight difference associated with a change from forged titanium to forged steel fan blades. For comparison, the Teledyne CAE 440-2 fan stage was selected for study, including the twelve configurations used in the previous cost reduction study. These configurations were the aspect ratios of 1.0, 2.0, 3.0 and 4.0 for the basic engine size, the half-thrust engine size and the twice-thrust engine size.

Similar generalized methods for estimating the machining and material costs of a forged steel blade were established. The assessment of the cost factors necessary to produce the steel blades yielded the following:

| | |
|---|---|
| Finishing of the airfoil surface | $ 6.75 per 10 square inches |
| Dovetail grinding | $ 3.00 per blade |
| Polishing | $ 1.25 per inch of blade height |
| Part shroud finishing, coating, heat treat, etc. | $ 7.50 per blade |
| Forging material | $ 1.95 per pound |

The blade/dovetail geometry and support discs were sized for the steel blades. Drawings were made of all twelve configurations. These drawings were used to calculate the finished weight, the forging weight and airfoil surface area for each blade. The results of these calculations are:

| | Aspect Ratio | Finished Weight (lbs) | Forging Weight (lbs) | Airfoil Surface Area ($in^2$) |
|---|---|---|---|---|
| Half-Thrust Size | 1.00 | 1.82 | 2.72 | 44.6 |
| | 2.00 | 0.45 | 0.68 | 22.3 |
| | 3.00 | 0.20 | 0.30 | 14.9 |
| | 4.00 | 0.11 | 0.17 | 11.1 |
| Basic Thrust Size | 1.00 | 5.14 | 7.71 | 88.9 |
| | 2.00 | 1.27 | 1.91 | 44.4 |
| | 3.00 | 0.57 | 0.85 | 29.7 |
| | 4.00 | 0.32 | 0.48 | 22.2 |
| Twice-Thrust Size | 1.00 | 14.53 | 21.80 | 178.0 |
| | 2.00 | 3.60 | 5.40 | 89.0 |
| | 3.00 | 1.61 | 2.41 | 59.4 |
| | 4.00 | 0.91 | 1.36 | 45.0 |

The method for estimating the machining and material costs for the support discs was developed for the previous cost reduction study. The disc material remained unchanged with only the configuration modified to support the heavier steel blades. The forging weights, finished weights, machining costs and material costs for each case are summarized in Figure 4.2.2-1.

The fan rotor weight versus aspect ratio for the forged steel blades is plotted in Figure 4.2.2-2. As with titanium blades, fan rotor weight decreases with increasing aspect ratio. For lower thrust (smaller engines), aspect ratios greater than two offer little improvement. Similarly, the fan rotor cost versus aspect ratio (Figure 4.2.2-3) shows optimum rotor costs. Actual rotor costs with steel blades are lower than those of a rotor with titanium blades.

As was the case for the titanium blades, fabrication limits of the leading and trailing edge radii cause compromises to the airfoils of the smaller blade sections, degrading performance of the steel fan blades identically to the titanium fan blades. Since the performance is the same for blades of either material, the only difference between the steel and titanium blades are the fabrication costs and the system flyaway cost impact due to the weight differences.

Using the Cost Analysis Summaries for the titanium fan blades and the forged steel fan blades (Figures 4.2.1-1 and 4.2.2-1, respectively) and the weight factors to assess system cost impact, the cost difference due to material selection has been determined. The rotor weight difference times the weight factor for each mission less the manufacturing cost difference gives the additional system cost for the use of steel blades. These values are:

| | | MMRPV System Cost Differential ($) | | UPT System Cost Differential ($) | |
|---|---|---|---|---|---|
| | | Less Part Span Shroud | With Part Span Shroud | Less Part Span Shroud | With Part Span Shroud |
| 0.5 | 1 | 789.54 | 684.54 | 1228.05 | 1123.05 |
| | 2 | 453.33 | 243.33 | 709.36 | 499.36 |
| | 3 | 303.23 | -11.77 | 473.78 | 158.78 |
| | 4 | 237.36 | -182.64 | 369.30 | -50.20 |
| 1.0 | 1 | 2553.80 | 2448.80 | 3998.32 | 3893.32 |
| | 2 | 1039.73 | 1099.73 | 2047.00 | 1837.00 |
| | 3 | 878.92 | 561.92 | 1371.72 | 1054.72 |
| | 4 | 662.84 | 242.84 | 1033.98 | 613.98 |
| 2.0 | 1 | 7213.19 | 7108.19 | 11,276.86 | 11,171.86 |
| | 2 | 3429.19 | 3219.19 | 5400.00 | 5190.00 |
| | 3 | 2318.73 | 2003.73 | 3635.04 | 3320.04 |
| | 4 | 1900.05 | 1480.05 | 2964.58 | 2544.58 |

| Scale Factor | Aspect Ratio | No. of Blades | Compressor Blades (Steel) | | | | | | | | Compressor Hub (Titanium) | | | | | | Compressor Rotor Assembly | | |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| | | | Material Weight | Material Cost | Blade Machining | | | | Finish Weight | Cost/Blade (Less Part Span Shroud) | Material Weight | Material Cost | Hub Machining | | Finish Weight | Cost/Hub | Total Weight | Total Cost Less Part Span Shroud | Total Cost With Part Span Shroud |
| | | | | | Dovetail Grinding | Airfoil Grinding | Leading and Trailing Edge | Part Span Shroud | | | | | Mach | Broach | | | | | |
| 0.5 | 1 | 14 | 2.72 | $5.31 | $3.00 | $69.40 | $5.25 | $7.50 | 1.82 | $82.96 | 47.60 | $266.60 | $17.07 | $21.00 | 12.49 | $304.67 | 32.97 | $1466.11 | $1571.11 |
| | 2 | 28 | 0.675 | 1.31 | | 34.50 | | | 0.45 | 44.06 | 26.93 | 150.80 | 10.40 | 42.00 | 5.63 | 203.20 | 18.23 | 1436.88 | 1646.88 |
| | 3 | 42 | 0.302 | 0.59 | | 23.07 | | | 0.20 | 31.91 | 21.45 | 120.12 | 8.60 | 63.00 | 3.70 | 191.72 | 12.14 | 1531.94 | 1846.94 |
| | 4 | 56 | 0.170 | 0.33 | | 17.34 | | | 0.113 | 25.92 | 18.10 | 101.40 | 7.19 | 84.00 | 3.31 | 192.59 | 9.64 | 1644.11 | 2064.11 |
| 1.0 | 1 | 14 | 7.71 | 15.04 | 3.00 | 137.78 | 7.50 | 7.50 | 5.14 | 163.32 | 103.92 | 581.90 | 34.20 | 21.00 | 33.67 | 637.10 | 105.67 | 2923.58 | 3028.58 |
| | 2 | 28 | 1.91 | 3.72 | | 69.80 | | | 1.27 | 84.02 | 58.54 | 327.80 | 20.70 | 42.00 | 15.93 | 390.50 | 51.53 | 2743.06 | 2953.06 |
| | 3 | 42 | 0.852 | 1.66 | | 46.10 | | | 0.568 | 58.26 | 46.03 | 257.8 | 17.20 | 63.00 | 10.48 | 338.00 | 34.34 | 2784.92 | 3099.92 |
| | 4 | 56 | 0.48 | 0.94 | | 34.72 | | | 0.320 | 46.16 | 38.79 | 317.20 | 14.40 | 84.00 | 9.13 | 315.60 | 27.05 | 2900.56 | 3320.56 |
| 2.0 | 1 | 14 | 21.8 | 42.5 | 3.00 | 277.53 | 10.50 | 7.50 | 14.53 | 333.53 | 227.0 | 1271.20 | 68.30 | 21.00 | 86.53 | 1360.50 | 289.90 | 6029.92 | 6134.92 |
| | 2 | 28 | 5.4 | 10.53 | | 138.17 | | | 3.60 | 162.20 | 127.5 | 718.90 | 41.40 | 42.00 | 42.29 | 802.30 | 143.09 | 5343.90 | 5553.90 |
| | 3 | 42 | 2.41 | 4.70 | | 92.37 | | | 1.61 | 110.57 | 98.9 | 553.90 | 34.40 | 63.00 | 27.84 | 651.30 | 95.38 | 5295.24 | 5610.24 |
| | 4 | 56 | 1.36 | 2.66 | | 69.38 | | | 0.91 | 85.54 | 84.4 | 472.60 | 28.70 | 84.00 | 25.29 | 585.30 | 76.25 | 5375.54 | 5795.54 |

Figure 4.2.2-1. Forged Steel Fan Blade Cost Analysis Summary.

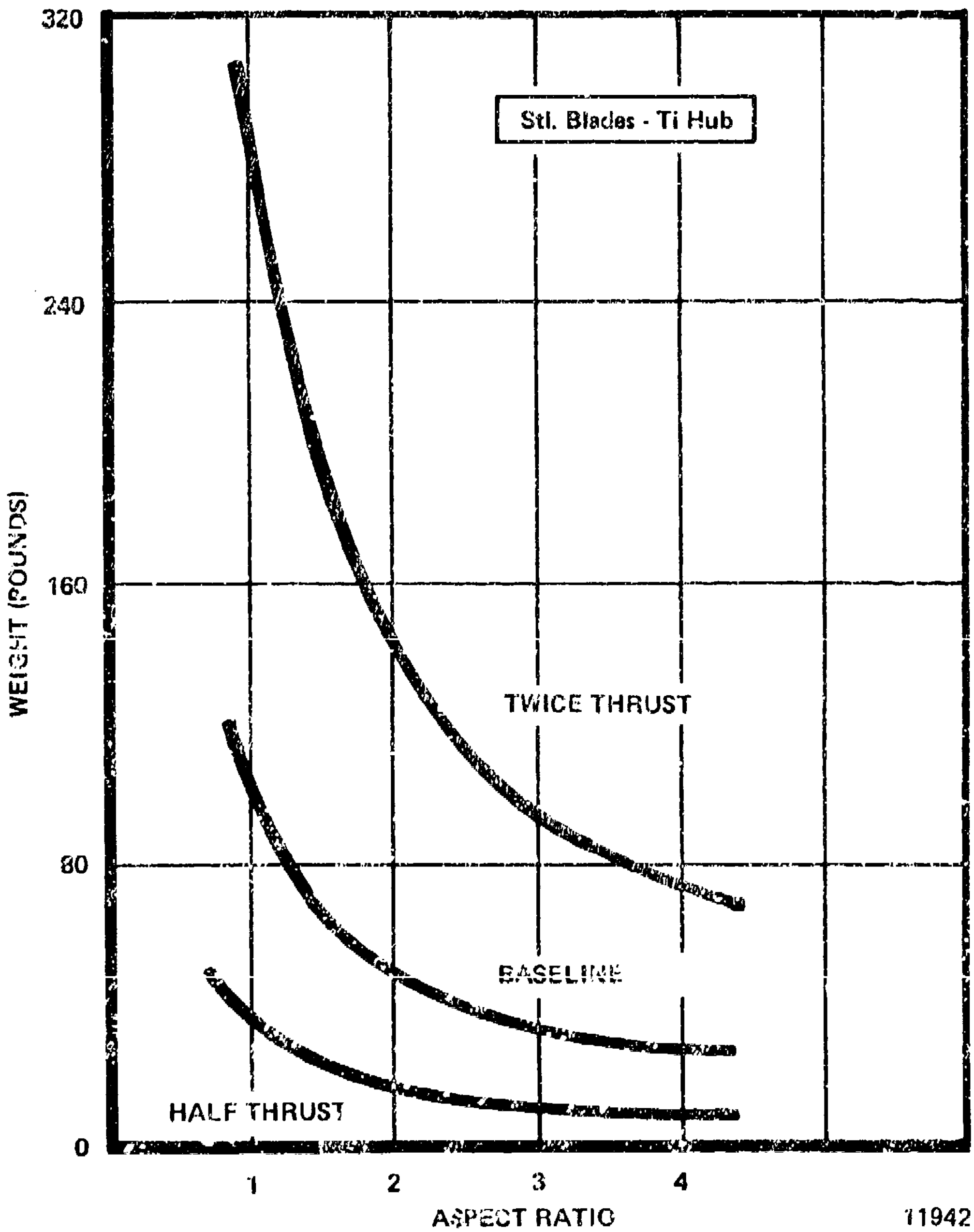


Figure 4.2.2-[illegible]. Fan Rotor Weight Versus Aspect Ratio - Steel Blades and Titanium Hub.

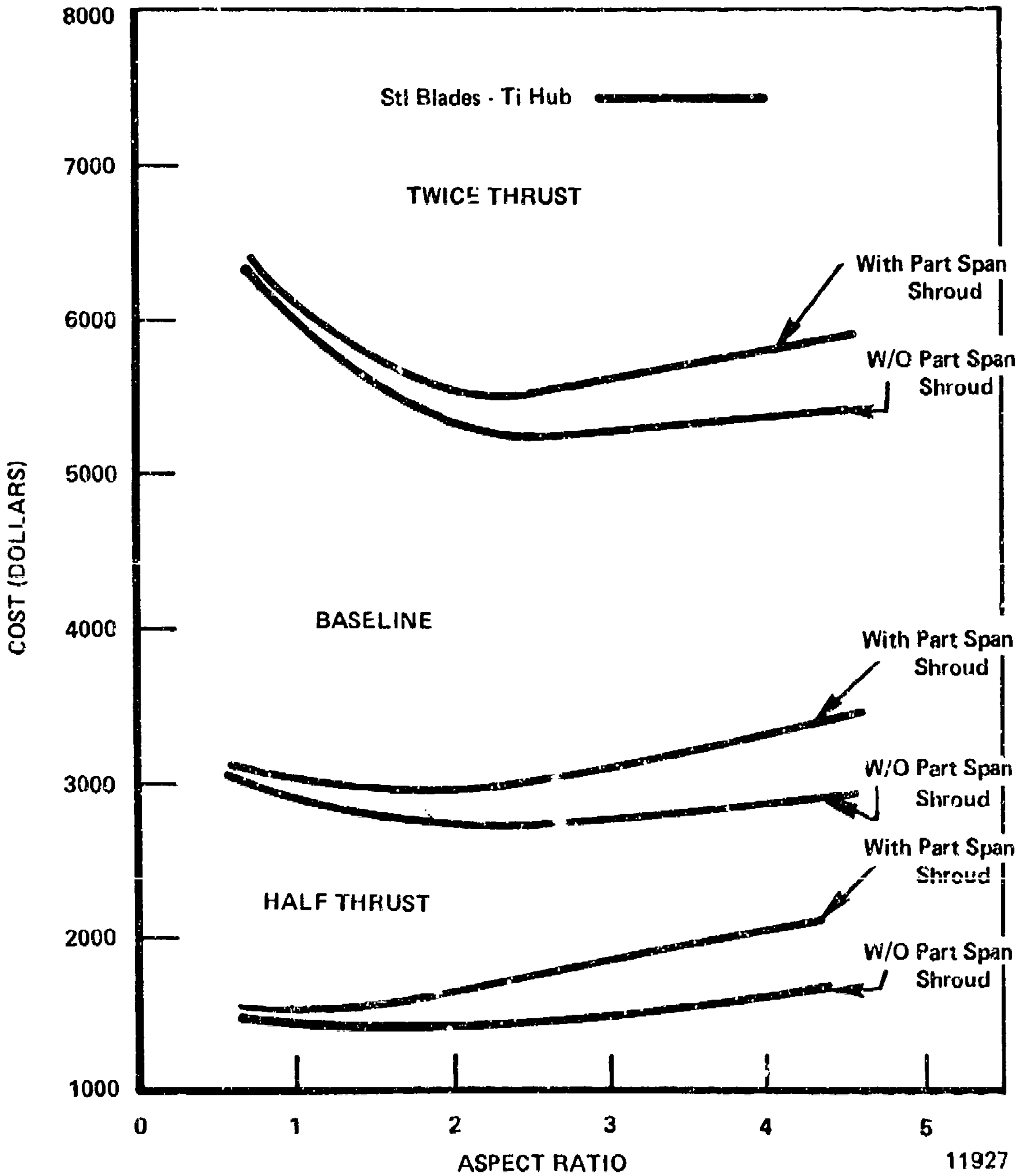


Figure 4.2.2-3. Fan Rotor Cost Versus Aspect Ratio - Steel Blades and Titanium Hub.

As shown, there are only three instances where the steel blades have less system cost than the titanium blades. These are for the small, high aspect ratio blades, represented in the table by the negative values. Aspect ratios must be selected with caution because of the severe performance degradation frequently incurred by these blades.

### 4.2.3 Cast Titanium Components

The objective of this study was to evaluate the cost and weight savings associated with using cast titanium engine components. The fan and compressor rotating components were selected as the most probable candidates to benefit from the proposed material and fabrication substitution because of their high manufacturing costs and low temperature operating environment. The advantages of using titanium alloys in these components are confirmed by previous results and comparison of Ti-6-4 alloy's strength, weight and temperature capability to other currently used compressor alloys.

Conventional forged/machined fan and axial compressor rotors require extensive precision machining to obtain the airfoil shapes and close tolerances resulting in costly hardware. By the use of investment cast titanium and integral bladed rotors, significant cost savings can be realized for both components. Additional weight savings for the axial rotor are possible because of the lower density of titanium compared to the current casting material, 17-4PH steel.

The justification for recommending cast titanium for these critical rotating components is based on the casting program conducted at Teledyne CAE on an axial compressor rotor and an axial compressor blade. These programs used 6Al-4V titanium alloy and demonstrated the capability of filling thin airfoil sections (0.008 to 0.012 inch leading and trailing edge thickness). The airfoils were cast slightly oversize, then chemically milled to size. The resulting surfaces and airfoil radii were suitable for engine aerodynamic evaluation. In addition, the benefits of hot isostatic pressing (HIP) were demonstrated as a means of removing shrinkage defects from the airfoil. Material properties testing up to 800°F indicated the suitability of these cast parts for rig and engine testing. The good ductility of 8 percent at room temperature was particularly significant. A comparison of data for wrought and cast HIP'ed titanium is shown in Figure 4.2.3-1. Although the cast material has slightly lower strength, a rotor burst margin of 1.25 is provided by the minimum properties of 110,000 psi tensile strength and 100,000 psi yield strength, determined from test specimens removed from sample castings.

The structural integrity of this complex, integrally bladed cast titanium rotor was demonstrated by a successful spin test to 110 percent of engine mechanical speed. The cast rotor is shown in Figure 4.2.3-2. The application of cast titanium was successfully demonstrated for the individual compressor blades as illustrated in Figure 4.2.3-3. These blades, of

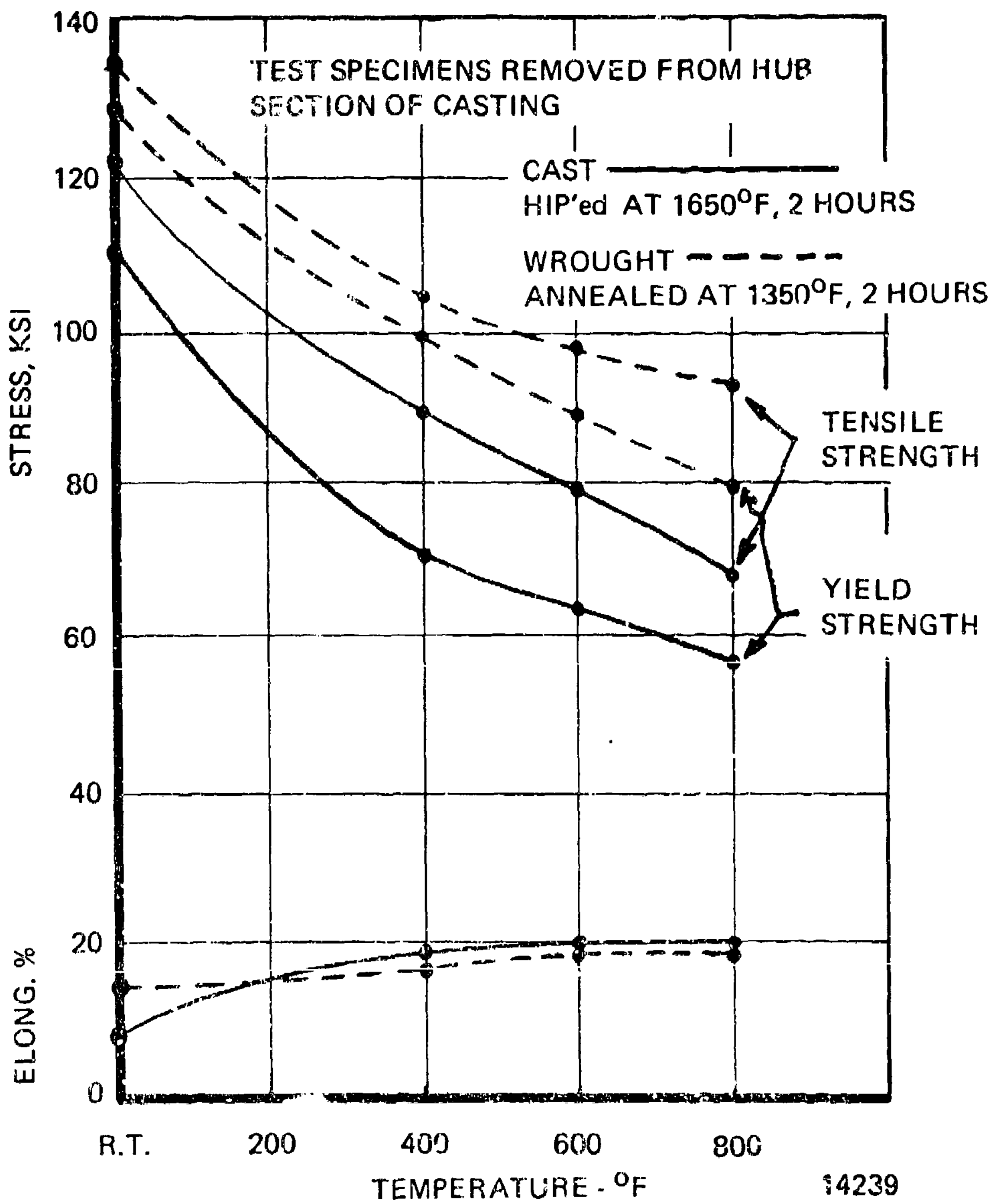


Figure 4.2.3-1. Tensile Properties Comparison of Cast and Wrought Titanium (6AL-4V).

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Figure 4.2.3-2. Investment Cast Titanium Axial Compressor Rotor.

which a dozen or more were cast for evaluation, are approximately the same size as the 455 fan blades.

With the feasibility of substituting cast titanium established, the fan and compressor rotors were subjected to the DTC costing procedures. The weights of the cast titanium components were calculated and compared to the baseline component weights. The axial compressor rotors showed a savings of 8.1 pounds when compared to the three stages of INCO 718 of the baseline configuration. Integrally cast fan blades and discs showed a weight reduction in the hub since material was no longer required for the dovetail attachments. This resulted in weight savings of 3.5 pounds for a single fan and 4.22 pounds for the two fan stages. Applying G & A and Profit to the DTC Estimates, and using the weight factors for the missions analyzed provides the following system flyaway cost reductions based on components chemically milled and hot isostatic pressed (HIP).

| Component | UPT (2 engines) | MMRPV |
|---|---|---|
| Cast Titanium Fan(s) | $ 10,855 | $ 6,872 |
| Cast Titanium Axial Compressors | $ 13,849 | $ 6,613 |

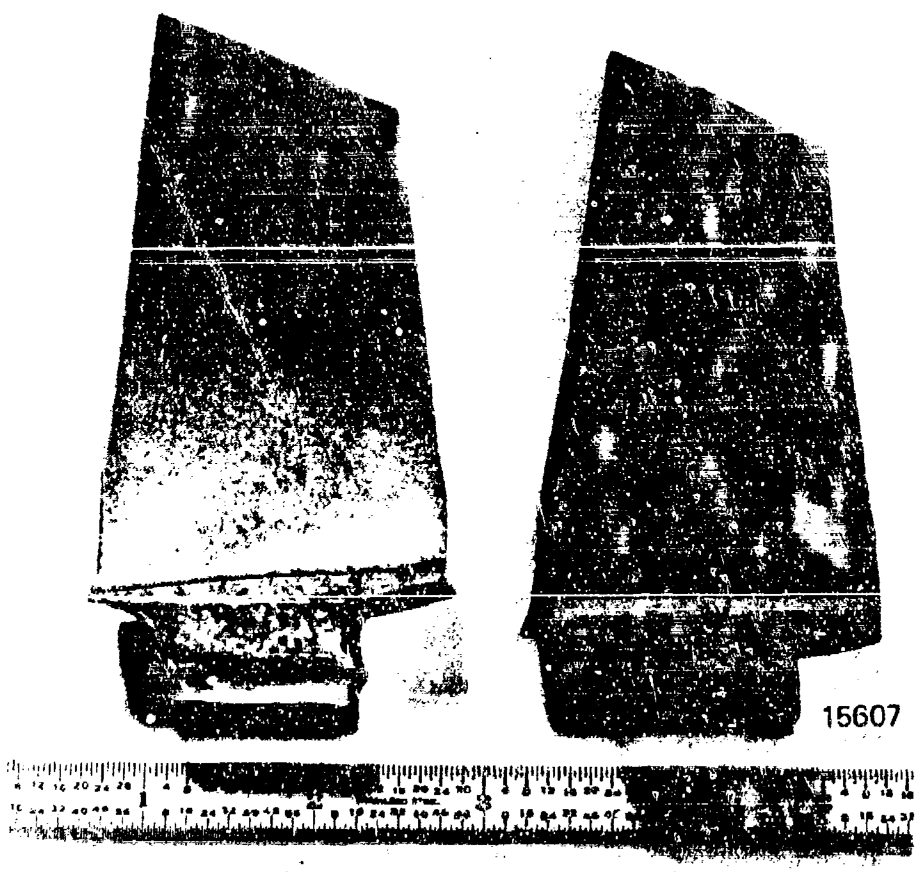


Figure 4.2.3-3. Investment Cast Titanium Compressor Blades.

4.2.4 Beryllium/Titanium Fan Power Shaft

The objective of this study was to determine the impact on engine cost and weight by using a beryllium/titanium alloy fan power shaft. Beryllium/titanium composites have excellent stiffness-to-weight ratios which encourage weight reduction studies. A Be/Ti composite of 70 percent beryllium (by volume) was selected to obtain an optimum stiffness-to-weight ratio for shafts. The volume percents chosen give a density of 0.0913 $lb/in^3$ and a Young's Modulus of 33.6 million psi. A preliminary design of a two-stage titanium fan and a two-stage steel power turbine was analyzed for shaft critical speeds and natural frequencies utilizing Teledyne CAE's Computer Program 02.074, the "Critical Speed Multiple-Shaft System". Shafts with varying diameters were evaluated to determine the minimum shaft size and resulting passage through the gas generator. The smallest shaft analyzed (1.4 inch outside diameter) has a free-mode critical speed of 33,837 rpm, providing a margin of greater than 50 percent on the low pressure shaft speed. The natural frequency of the same shaft computed at 22,000 rpm equals 41,182 rpm for a support stiffness of $10^6$ lbs/in. Figure 4.2.4-1 is a plot of frequencies versus support spring rates for the recommended shaft. The natural frequencies of the shaft can be altered to avoid resonant conditions during detail design by judicious selection of the bearing support stiffnesses. Figure 4.2.4-1 shows that the 1.4 inch outside diameter shaft will have a margin on bending frequency of 10 percent with the high pressure shaft speed with both bearing spring rates equal to 640,000 lbs/in.

The preliminary analysis indicates the feasibility of obtaining satisfactory dynamic performance using a beryllium/titanium composite shaft. With this basic assumption, the engine components were redesigned to realize the greatest advantage possible by using a smaller diameter shaft. The gas generator discs were analyzed with the bore diameters decreased by 0.8 inch and the cross-sections recontoured to provide identical burst margins to the current design.

The proposed Be/Ti composite is basically a product of powder metallurgy extrusion. The fabrication techniques involved are relatively uncomplicated. Beryllium and titanium powders are mixed together, canned in a jacket, and extruded. The extrusion process converts the beryllium and titanium powders into micro fibers which are diffusion-bonded under the influence of the extrusion temperature and pressure. Metallurgical examination has shown that excellent diffusion bonds, devoid of major concentrations of titanium beryllide reaction products, are the result of this process.

The composite is easily produced with a monolithic titanium alloy surface "skin" which is desirable for the machining of threads and splines on the external surface. This composition is shown in Figure 4.2.4-2, and the canning process is presented in Figure 4.2.4-3. The latter figure represents the method which would be used to provide a Be/Ti shaft with

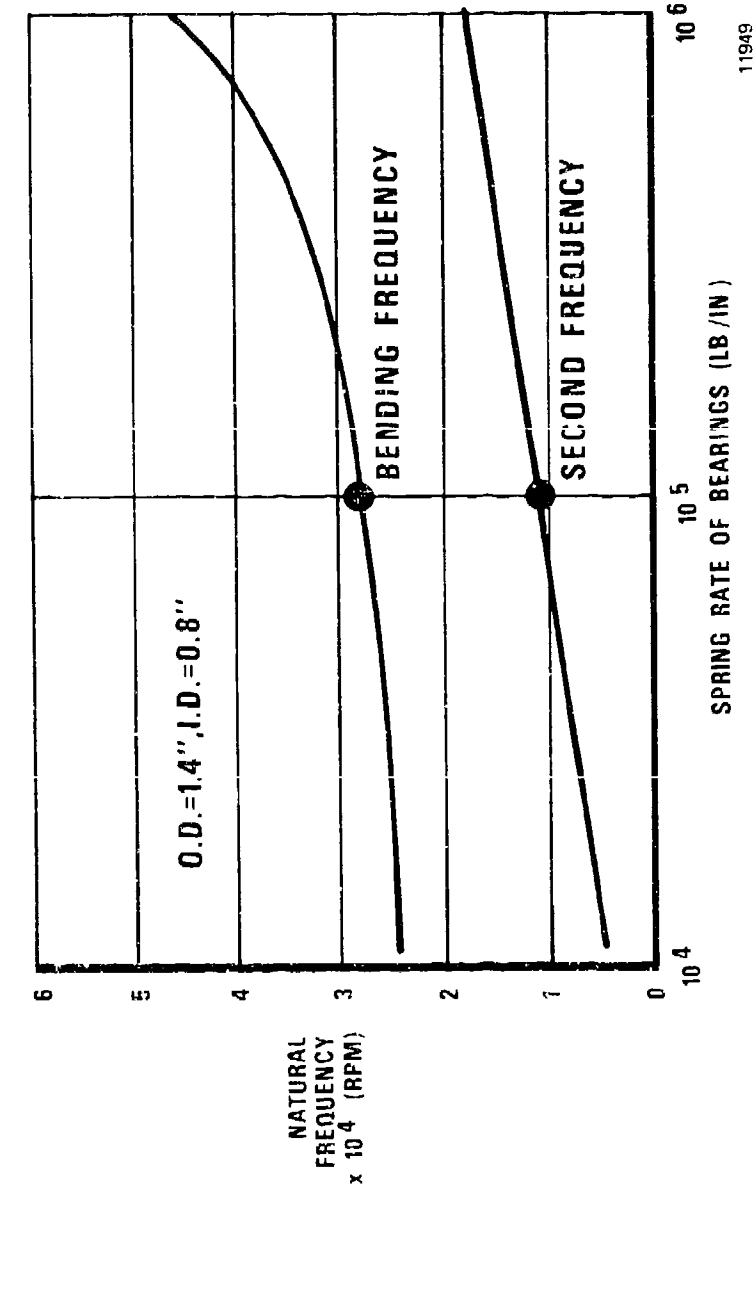


Figure 4.2.4-1. Be/Ti Low Pressure Shaft - Natural Frequencies Versus Support Spring Rates at 22,000 rpm.

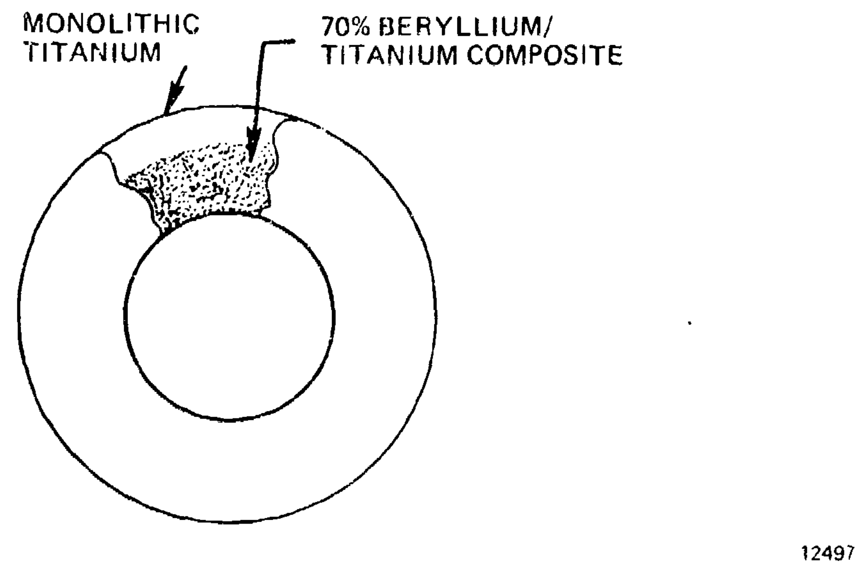


Figure 4.2.4-2. Schematic Presentation of Shaft Composition.

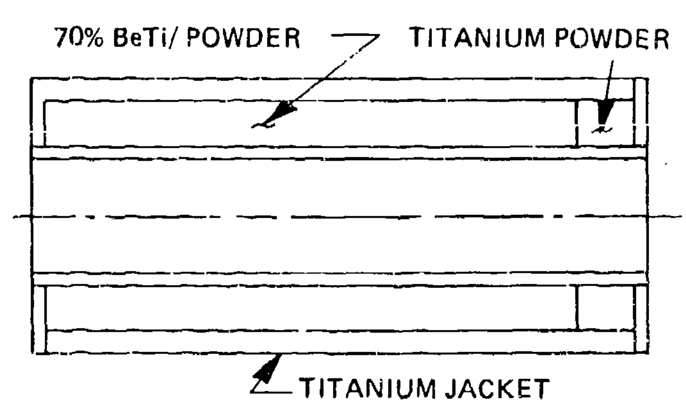


Figure 4.2.4-3. Canning Process for Required Material Composition.

titanium ends to reduce machining hazards. This construction (Figure 4.2.4-4) requires machining of only the titanium material to finish machine the shaft. The only special handling required would be for gun drilling the bore of the shaft, if required for balancing accuracy. With this construction, the wear and gall resistance can be assured by use of graphite varnish on the cold end and the Tiduran process on the hot end.

The favorable results of the feasibility studies of both the structural integrity and fabrication technique for the Be/Ti composite shaft prompted a cost and weight comparison, using similar construction of all the involved components. The cost analysis included material costs determined from vendor furnished projects (Figure 4.2.4-5) and machining costs as determined by the special handling required for the machining of the Be/Ti composite. Application of this cost reduction to the derivative engines for the UPT and MMRPV missions provide the following system fly-away cost payoffs:

| | | | |
|---|---|---|---|
| UPT | (two engines) | - | \$5,802 |
| MMRPV | | - | \$2,291 |

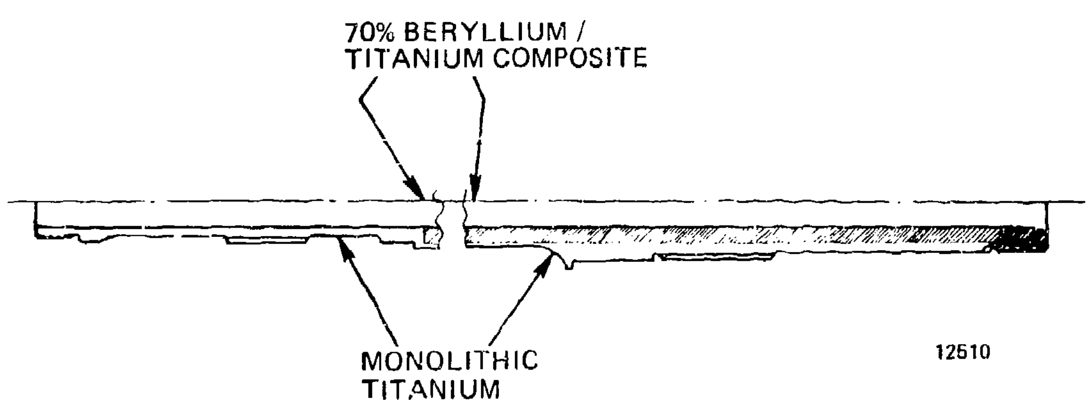


Figure 4 2.4-4. Be/Ti Configuration and Composition.

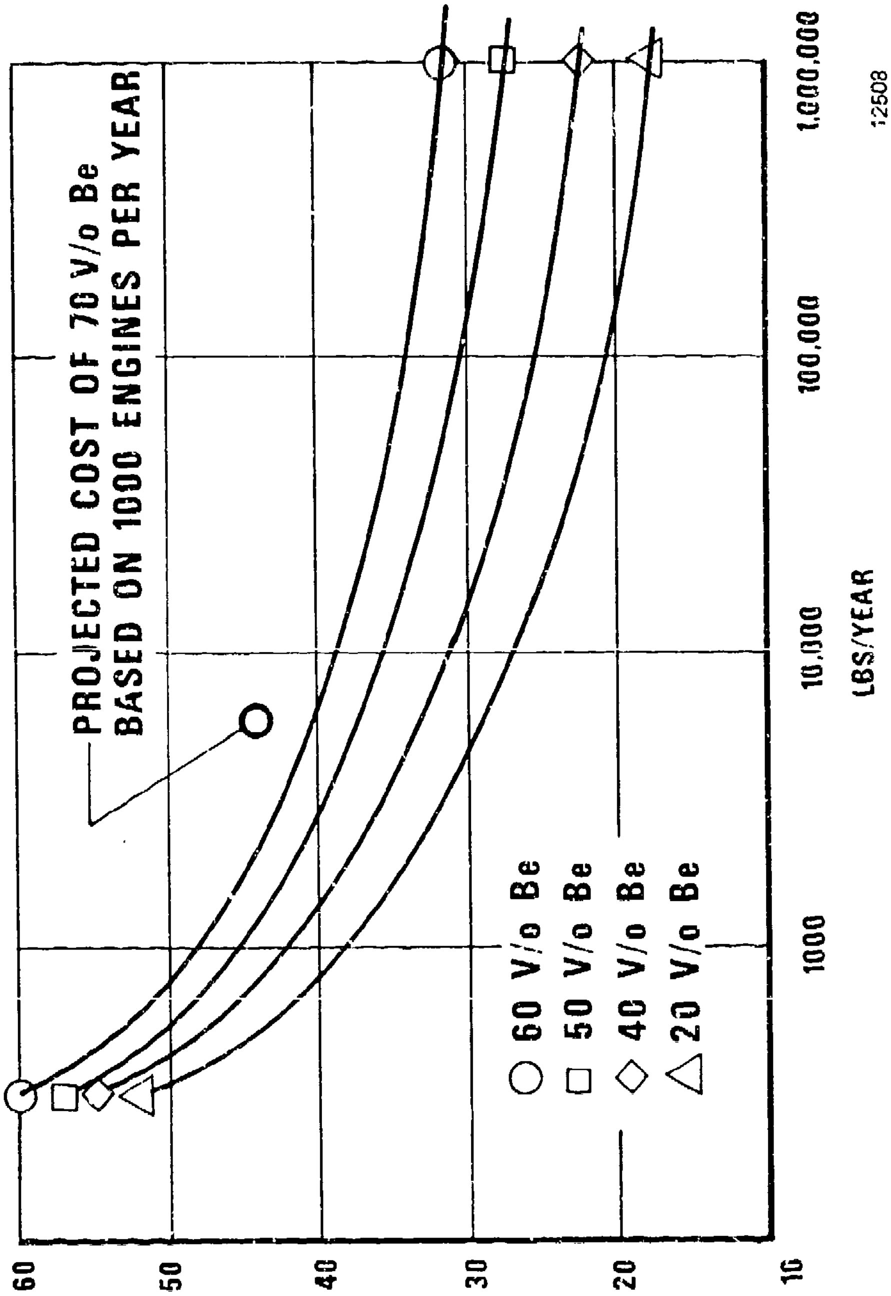


Figure 4.2.4-5. Be/Ti Price Projections Data (Furnished by Brush Wellman).

4.2.5 Brazed Axial Compressor

The objective of this study was to reduce costs by using formed-to-finish size titanium axial compressor blades brazed into a one-piece drum structure.

The concept of brazing finished titanium compressor blades on the axial compressor discs has been presented in a preliminary design layout (Figure 4.2.5-1). A drawing of the first stage blade (Figure 4.2.5-2) was made to determine the structural feasibility, manufacturing the airfoil shapes finished to size in titanium and to determine the resultant tolerances for each of the manufacturing processes. The processes considered were precision forgings, precision castings, and powder metal consolidation. Our present machined tolerances can provide contour tolerance bands of plus or minus two thousandths of an inch. The precision cast blades, chemically milled to remove the alpha case, along with the powder metal fabrication, can provide the same contour tolerance bands. The precision forged blades can provide contour tolerance bands of plus or minus three thousandths of an inch. The aerodynamic performance variation due to such a minuscule tolerance deviation is not within the limits of practical determination and is considered minimal. In actual practice, the normal tolerance distribution would never provide the extremes of the contour tolerance band. For this reason, all of the manufacturing processes were considered to provide equivalent aerodynamic performance.

The foregoing tolerance determination provided the basis for establishing the feasibility of brazing the titanium blades onto the drum. With the airfoil contour tolerance set at plus or minus 0.003 inch and the punched airfoil contours in the rings at tolerances of plus or minus 0.002 inch, the brazing gap will be from line to line to 0.014 inch maximum. Contacts were established with vendors experienced in titanium brazing and with braze alloy sources to confirm the brazing capability using our components. Once the surface contamination of titanium is overcome to afford good wetting, titanium is easily brazed. There are many braze alloys available for titanium brazing. However, to maintain good ductility and wide gap capability, an alloy such as AMDRY 692 should be utilized. This is a silver-palladium-aluminum alloy which uses the palladium to form a phase barrier against the diffusion of titanium and silver. AMDRY 692 braze alloy has high ductility and wide gap (up to 0.030 inch) brazing capability. It is also a low temperature brazing alloy from 1500°F to 1550°F and has a shear strength of 35,000 psi.

The manufacturing concept described above is feasible; however, sample braze joints should be evaluated to confirm strength characteristics.

A preliminary structural analysis was conducted on the proposed design. The blade stresses, previously calculated for both cast 17-4 steel and INCO 718 alloy, were not calculated because the strength-to-density ratio of titanium is greater than the steels, thereby providing greater safety margin than the present design.

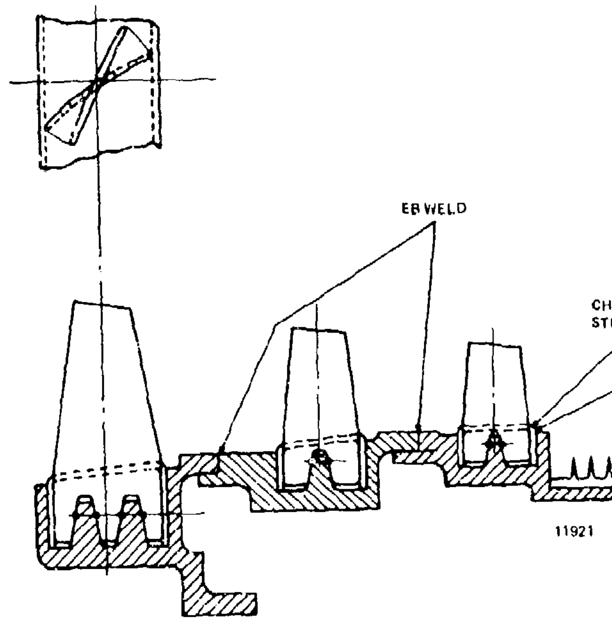


Figure 4.2.5-1. Preliminary Design Layout of Three-Stage Axial Rotor.

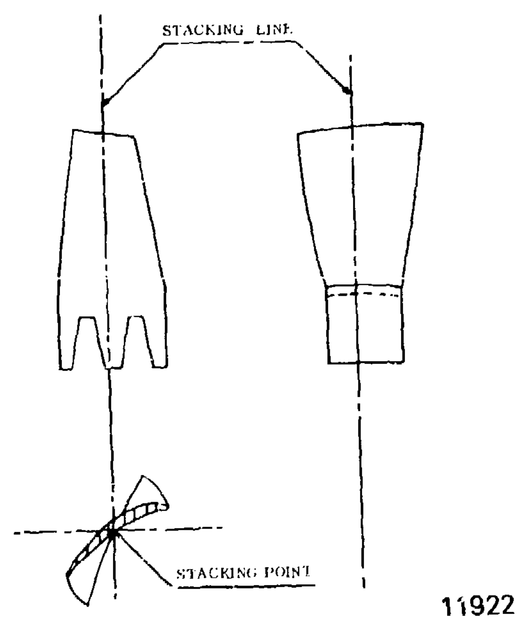


Figure 4.2.5-2. Rotor Blade.

The analysis focused on the blade attachment stresses based on the assumption that all of the braze area shares the blade centrifugal load. In fact, the braze in the sheet metal ring carries the load until it yields enough to permit the remaining areas to pick up a share. The interaction between the brazes in the sheet metal ring and in the disc should also be investigated thoroughly during the design and evaluation program.

The gas loadings were reacted by the braze in the sheet metal ring. An equivalent moment of inertia using the braze shear area was calculated at 0.000636 $in^4$ about the minimum axis, and 0.02035 $in^4$ about the maximum axis. A ring thickness of 0.050 inch was used. The resultant calculated gas bending braze stress is 2032 psi.

Figure 4.2.5-3 gives a summary of the blade attachment stresses. The combined stress of 10,182 psi considers 100 percent of the braze area to be effective. Braze is rarely 100 percent effective and the allowable stress will not be known until the brazing alloy has been evaluated. However, based on the 10,182 psi stress, a safety factor of 1.47 can be achieved with only 50 percent braze effectivity, if the braze material allowable stress is on the order of 30,000 psi.

Total area of braze ($in^2$) = 0.3037

Blade pull (titanium, 100% Speed) (lb) = 2475

Stress P/A (psi) = 8150 (for equal load sharing throughout)

---

Gas bending moments (in/lb) tangential = 6.33

(in/lb) axial = 4.15

Equivalent moment of inertia in shear of the braze in the sheet metal sleeve (0.050 in thick)

Minimum axis ($in^4$) = 0.000636

Maximum axis ($in^4$) = 0.02035

Assuming braze in sleeve resists all gas bending loads.

Stress (shear in braze at trailing edge) (psi) = 2032

---

Total stress in brazed joint (psi) = 2032 + 8150 = 10,182

Figure 4.2.5-3. Stress Summary - Brazed First Stage Blade on Ring Disc.

The rotor drum was analyzed to determine the maximum bore stresses of the rings comprising the three axial stages of the compressor. These stresses, summarized below as percentages of yield for comparison between the titanium rotor and the individual INCO 718 discs, are significantly lower than those of the current disc design.

| | Titanium Rotor | | INCO 718 Discs | |
|---|---|---|---|---|
| | Bore Stress (psi) | % of Yield | Bore Stress (psi) | % of Yield |
| First Stage | 80,000 | 69 | 112,000 | 76 |
| Second Stage | 69,000 | 69 | 104,000 | 72 |
| Third Stage | 60,500 | 69 | 107,000 | 75 |

The preliminary analysis shows that the brazed axial compressor concept is viable in terms of structural integrity.

A design layout was made which encompasses the mechanical design features selected to contribute to the cost reduction of the axial compressor section. One such feature is the one-piece drum construction, fabricated by EB welding the internally finished rings together. This rotor can subsequently be finish machined by crush-grinding the external diameters to provide close tolerance dimensions for the brazing surfaces and flowpath wall through use of a rapid, low cost material removal method. A comparison with the baseline engine configuration shows that it is possible to eliminate the compressor labyrinth seals and inner stator rings by cantilevering the stator vanes. Recirculation leakage is eliminated in two stages, and the rotating inner flowpath wall tends to eliminate vortices in the end wall stall region which are common to stators of this type. In addition, the one-piece rotor assembly requires less balancing and assembly time than the individual rotors and seals because the integral three-stage rotor only needs to be calibrated once on the balancing equipment, while the individual rotors must be calibrated three times.

Costs were estimated for the integral drum rotor compressor using the DTC costing procedures, and then compared to the baseline configuration. The weight of this configuration, when compared to the baseline weight, showed a savings of 8.48 pounds. Applying G & A and Profit to the DTC Estimates, and using the weight factors established from mission analyses, yields the following system cost reductions:

| | | | |
|---|---|---|---|
| UPT | (two engines) | - | $9,350 |
| MMRPV | | - | $4,349 |

These substantial savings are not as great as those from cast titanium axial compressors ($13,849 and $6,613, respectively) but are certainly worthy of consideration as alternatives.

### 4.2.6 Hybrid Radial Compressor Diffuser

The objective of this study was to reduce the manufacturing cost and improve performance of the radial compressor diffuser by using a single mixed-flow diffuser instead of the separate radial and axial diffuser stator rows. The proposed mixed-flow diffuser was designed to the requirements of the baseline engine. The configuration provides a continuous diffusing channel from inlet to exit. The area distribution differs from that presently employed in the diffuser for the baseline engine, which has a radial diffuser coupled by a constant area elbow to an axial diffuser.

A cost analyses of the hybrid radial diffuser was made and compared to baseline configuration (radial and axial diffusers). The cost analyses for both types of diffusers were performed using the same fabrication criteria. Processes and machining were identical where the diffusers had common features, allowing the comparison to reflect the true cost differential due to the elimination of the attendant machining requirements for one part.

The hybrid radial diffuser cost estimate was compared to the baseline engine costs, and then G & A and Profit were applied to the difference. The one percent improvement in SFC was expressed in lbs/hr/lb for each of the engines used in the UPT and MMRPV missions, and the respective values of SFC in airframe flyaway cost were applied. The system flyaway cost reductions are in order of 1 - 6 thousand dollars, depending on the complexity of the aircraft of interest.

### 4.2.7 Infusion Cooled Vaporizer Plate Combustor

The objective of this study was to reduce the cost of the baseline engine combustor by eliminating fuel nozzles, film cooling strips, corrugated strips, and numerous machined holes for distribution of air flow. The chamber liner is fabricated from porous sintered wire mesh to provide a controllable film for infusion cooling. The injection system is annular, thus eliminating numerous parts associated with the fuel vaporizer pipes.

Three-layer Rigimesh, 12 x 64 mesh, is specified for the liner because the airflow pressure drop characteristics are appropriate for this application. A $\pm$ 12.5 percent tolerance band on the nominal airflow of 40 scfm/ft$^2$, established to minimize processing costs, results in the average liner metal temperature variation shown in Figure 4.2.7-1. Since the liner pressure loading is minimal (7 psi $\Delta P$ at design point), other structural considerations and/or oxidation resistance versus required life will govern the range of allowable metal temperature and the selection of the Rigimesh type base material.

The vaporizer plate effective flow area, sized to provide the combustion air required at design point, eliminates the need for auxiliary baseplate holes and associated life problems, thus minimizing fabrication and system operational costs.

The vaporizer plate length, determined by a heat balance across the surface area of the plate, was arbitrarily reduced by 25 percent to account for fuel film turbulence and wave interaction with the primary combustion air. These secondary effects are difficult to accurately consider in the heat balance, but would enhance the heat input to the fuel. The vaporizer plate length and the degree of vaporization at various operating conditions and locations will be verified by test.

The infusion cooled vaporizer combustor provides the potential for removing the axial diffuser vanes from the radial compressor diffuser. The infusion cooled liner readily accommodates the residual swirl in the air entering the combustor. The residual swirl will be advantageous entering the HP turbine inlet nozzle, allowing at least 10 percent of the turbine inlet nozzle vanes to be removed. The increased pressure loss of the swirling air entering the combustor is offset by the decreased pressure loss caused by removing the axial diffuser. The total effect on the engine is to maintain the same level of performance with fewer parts.

A detailed cost analysis was performed to determine the comparative cost of the infusion cooled vaporizer plate combustor with the baseline engine configuration. This cost comparison is summarized in Figure 4.2.7-2. As expected, elimination of the fuel nozzles and the cooling strips, and the multiplicity of holes in the combustor liners drastically lowers the number of labor hours required for manufacturing. Although the material cost increases by over $1700 for all items (primarily because of the preformed porous material), the total savings attributed to this configuration when G & A and Profit are applied to the costs is $3,776.00. Additionally, there is a weight reduction of 5.4 pounds for the proposed configuration. When the mission-established weighting factors are applied, the system flyaway cost reductions are:

| | | | |
|---|---|---|---|
| UPT | (two engines) | - | $8,778 |
| MMRPV | | - | $4,181 |

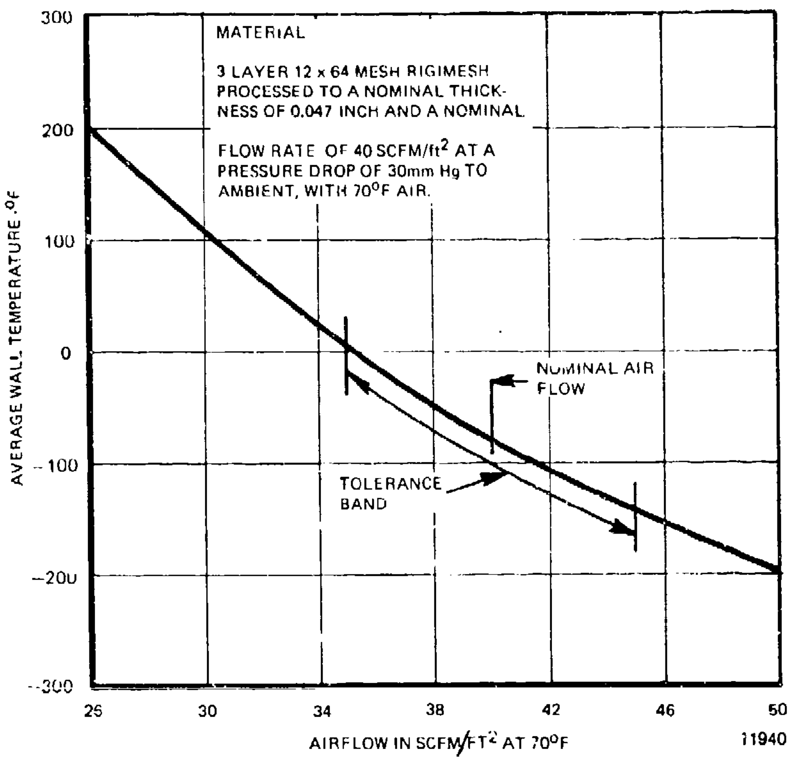


Figure 4.2.7-1. Effect of Tolerance Band on Liner Temperature.

| | Labor Cost | Material Cost | Total Cost |
|---|---|---|---|
| Baseline Combustor | $ 3374 | $ 259 | $ 3633 |
| Baseline Manifold | 1725 | 58 | 1783 |
| Axial Diffuser Vanes | 98 | 352 | 450 |
| Turbine Inlet Nozzle | 30 | 2914 | 2944 |
| Total-Baseline Engine Configuration | 5227 | 3583 | 6810 |
| Infusion Cooled Vaporizer Plate Combustor | $ 541 | $2708 | $ 3249 |
| New Manifold | 151 | 14 | 165 |
| Turbine Inlet Nozzle | 27 | 2622 | 2649 |
| Total - Reduced Cost Configuration | 719 | 5344 | 6063 |
| Differential Cost | -$4508 | +$1761 | -$2747 |

Figure 4.2.7-2. Cost Comparison - Infusion Cooled Vaporizer Plate Combustor Versus Baseline Engine Configuration.

### 4.2.8 Ceramic Components

The objective of this study was to achieve reduced engine weight and cost by implementing ceramic materials for engine components, particularly where cooling air is required for hot section metal components.

High engine temperatures require extensive cooling of metal components in the combustor, turbine and exhaust nozzle sections. Cooling of metal parts is very costly because the cooling techniques require complex coring in investment castings, small intricate hole patterns in sheet metal structures, oxidation resistant coatings, and the use of a significant percentage of engine airflow. Ceramics eliminate or reduce the need for cooling and offer significant cost reductions for the affected components and minimize the use of engine airflow, resutling in improved performance. The high and low pressure turbine inlet nozzles and the front transition section of the combustor shell were selected for analysis because of Teledyne CAE experience with static ceramic structures. The strength advantages of ceramics over metals at increasing temperatures are shown by the comparison in Figure 4.2.8-1. In addition, ceramics have low density and exhibit good oxidation resistance to 2500°F, thereby offering weight reduction and eliminating the need for a coating. Brittleness, their primary disadvantage, can be minimized by innovative design and by using techniques such as three-dimensional finite element structural analysis and fracture mechanics.

Candidate ceramic materials for these components include both reaction-bonded silicon nitride and silicon carbide. Hot pressed ceramics are not being considered because of their fabrication limitations. The cost advantage of using injection molding techniques to provide a turbine inlet nozzle configuration with nozzle vanes integral with the shroud (requiring only a minimum of machining on the flanges) is readily apparent. Fabrication of a ceramic high pressure turbine inlet nozzle provides a 50 percent cost reduction compared to the metal structure (based on supplier production cost estimates).

Teledyne CAE actively pursued the evaluation of ceramics for the last four years to develop both material characterization data and design analysis techniques. Material studies have addressed thermal shock and oxidation resistance of test specimens up to 2500°F. Cascade rig testing of nozzle vanes has also been conducted to the same temperature. Because of the brittle nature of ceramics, ballistic impact testing was conducted on nozzle vanes at engine velocity to determine their resistance to damage by foreign material.

The evaluation of ceramic materials components and design analysis techniques culminated in the running, under USAF sponsorship, of a full-round turbine inlet nozzle for 51 minutes at high temperatures in an ATEGG 440-2 gas generator.

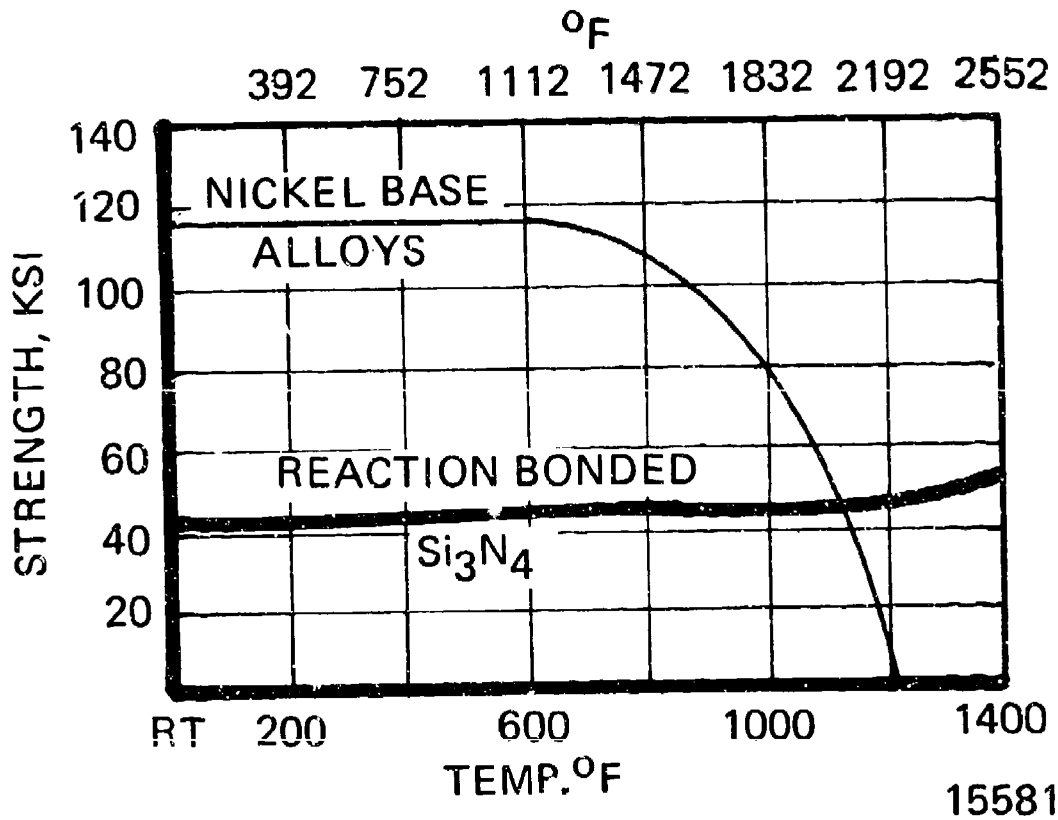


Figure 4.2.8-1. Strength Comparison of Ceramics ($Si_3 N_4$) and Superalloys.

Teledyne CAE is confident that progress in using these materials is rapidly leading to their practical application in gas turbine engines.

The use of ceramic materials for the HP and LP nozzles and the combustor transition duct is feasible at this time. Ceramic components were deemed not applicable to the UPT manned mission without first demonstrating their structural integrity. The application of the cost and weight benefits of ceramics for the MMRPV engine is considered feasible and provides the following system flyaway cost reductions for the components listed:

| | |
|---|---|
| Ceramic HP Nozzle | $5517 |
| Ceramic LP Nozzles | $2692 |
| Ceramic Combustor Transition Duct | $ 221 |

### 4.2.9 Powder Metal Components

The objective of this study was to examine powder metal consolidation methods and the attendant mechanical properties to compare the resulting costs and weights with current component fabrication methods. The consolidated powder metal approach to component fabrication offers the possibility of forming to near-finished shape, thus producing low-cost structures. This feature of powder metal to minimize labor is a major contributor to cost reduction. In this investigation, powder metal turbine blades were compared to cast blading, itself a formed-to-shape method, to determine the potential for cost reduction. Also considered was the use of sintered and HIP'ed powder metal discs as a replacement for forged discs. Hot isostatic pressing (HIP) assists in densification of the powder and the achievement of desired material properties.

The mechanical properties developed in the concurrent APSI effort are shown in Figure 4.2.9-1. In contrast to IN-100, the INCO 792 alloy is used in the heat treated condition. This heat treat cycle offers the opportunity to optimize grain size for the best combination of high temperature mechanical properties. A trial blade fabrication is shown in Figure 4.2.9-2.

Promising results with the turbine blades prompted an investigation to determine if a turbine disc would yield similar results. Use of the HIP process with either a metal or ceramic mold also allows fabrication of a turbine rotor disc to a near net shape - reducing machining by about 30 percent. A current Air Force Materials Laboratory Program addresses the manufacturing development of powder metal discs by hot isostatic pressing.

INCO 792 SINTERED AND HIP POWDER METAL*

| | RT | 1200°F | 1500°F |
|---|---|---|---|
| TENSILE ULTIMATE - KSI | 221.0 | 205 | 141.0 |
| TENSILE 0.2% YIELD - KSI | 161.0 | 153 | 132.0 |
| ELONGATION - percent | 11.4 | 10 | 5.3 |
| REDUCTION AREA - percent | 10.5 | 11 | 8.0 |
| | LOAD (KSI) | TEMP (°F) | TIME (HRS.) |
| STRESS-RUPTURE | 22.0<br>65.0<br>150.0 | 1800<br>1500<br>1200 | 32<br>70+<br>400+ |
| *HEAT TREAT 2050°F/2Hr. OQ, 1450°F/16 Hr. AC, 1250°F/16 Hr. AC. | | | |

| IN-100** | | | |
|---|---|---|---|
| | RT | 1200°F | 1500°F |
| TENSILE ULTIMATE - KSI | 170 | 165 | 143 |
| TENSILE 0.2% YIELD - KSI | 127 | 127 | 110 |
| ENONGATION - percent | 10 | 8 | 8 |
| | LOAD (KSI) | TEMP (°F) | TIME (HRS.) |
| STRESS-RUPTURE | 22.0<br>65.0 | 1800<br>1500 | 8<br>40 |

**COLD PRESSED AND SINTERED

Figure 4.2.9-1. Preliminary Mechanical Properties Comparison - INCO 792 Versus IN-100.

The use of powder metal for the ATECG high pressure turbine disc, HIP'ed to the configuration shown in Figure 4.2.9-3, would provide a 30 percent cost reduction from the current design. Since powder metal alloys such as Rene' 95 and Inconel 792 have significantly higher strength than the presently used conventional Waspalloy forging (Figure 4.2.9-4), the weight of the disc can also be reduced through redesign.

Detailed cost analyses of the HP and LP discs and blades have been conducted. Based on the cost and weight analyses of the powder metal components, their application in the UPT and MMRPV derivatives will produce the following system flyaway cost reductions:

| Component | UPT (two engines) | MMRPV |
|---|---|---|
| P/M LP Blades | $1340 | $ 436 |
| P/M LP Hubs | $2444 | $ 553 |
| P/M HP Blades | $2090 | $1045 |
| P/M HP Hubs | $1866 | $ 773 |

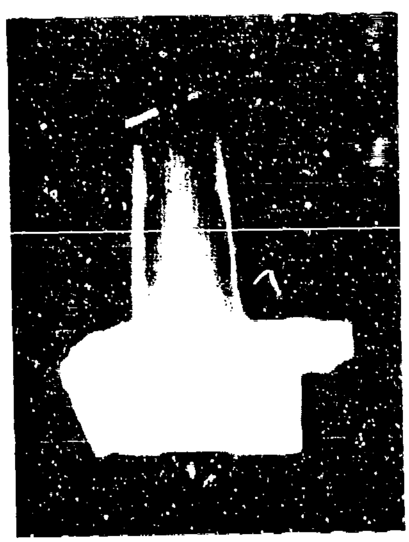

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Figure 4.2.9-2. Sintered and Hot-Isostatic-Pressed Powdered Metal Turbine Blade Made From INCO 792 (Radiographic Film).

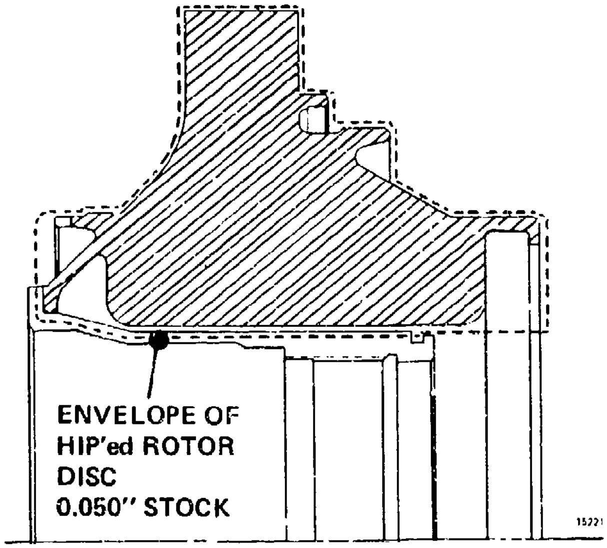



Figure 4.2.9-3. Powder Metal Disc Outline (Dashed).

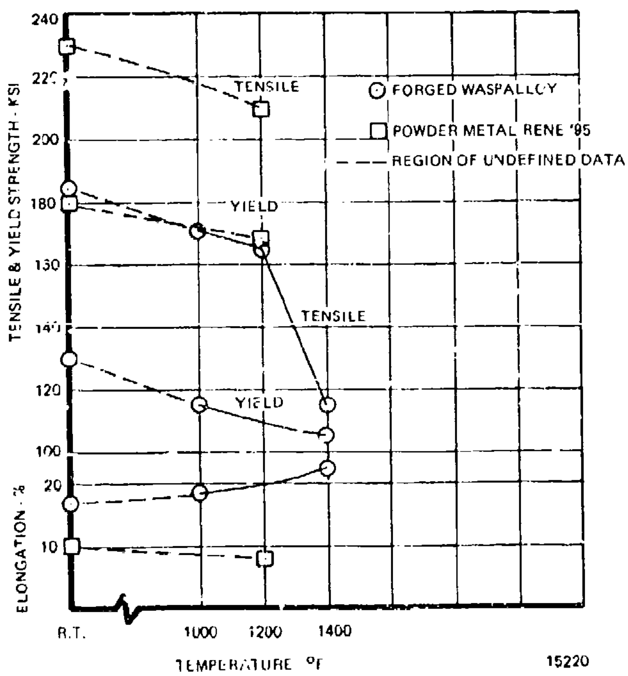



Figure 4.2.9-4. Tensile Properties of Forged Waspalloy and Powder Metal Rene' 95.

### 4.2.10 Elimination of LP Turbine Inlet Nozzle

The objective of this study was to evaluate the possibility of eliminating the low pressure turbine first stage nozzle to provide a cost advantage by use of counter-rotating high and low pressure shafts. Although complete elimination of the nozzle is desirable, it was considered too ambitious an undertaking except for an engine with a single primary point of operation. The off-design point performance tradeoffs over the operating ranges defined by mission analyses were determined to be too numerous to handle within the scope of these study programs. Therefore, this study used counter-rotating high pressure and low pressure spools to utilize the discharge swirl of the HP turbine section to unload the LP turbine nozzle. The net result is that fewer turbine inlet vanes are required, turning losses are reduced, and a corresponding increase in aerodynamic performance is achieved.

A preliminary velocity diagram analysis was conducted to investigate those factors most influential in governing turbine performance. These included: trailing edge thickness, tip clearance, pitch-chord ratio, blade-loading diagram, inlet swirl, and exit swirl.

Inlet swirl was determined to be beneficial, within limits, to unload the LP turbine inlet nozzle vanes.

Counter-rotation of the low pressure turbine relative to the gas generator turbine allows effective utilization of gas generator turbine gas exit swirl, thereby reducing the gas turning required in the LP turbine inlet nozzle as illustrated in Figure 4.2.10-1. This reduction in turning unloads the nozzle vanes, thus allowing a large reduction in the number of vanes for equivalent turning losses. In the design of one LP turbine, the design value of gas generator exit swirl, approximately 20 degrees, was employed to reduce the number of turbine inlet nozzle vanes. Later, the number of vanes was increased slightly to minimize a common multiple frequency problem. In addition, less complex airfoil shapes are achieved because of the reduced gas turning angle (Figure 4.2.10-1). This simplifies vane manufacturing by being able to maintain a constant thickness over most of the vane chord.

In translating these advantages to cost reduction, both castings and fabrications should be considered because the simpler airfoils are economical to fabricate. As shown in Figure 4.2.10-2, castings are cheaper to manufacture, unless the scrap rate becomes larger than usual. Counter-rotation yields cost and weight reductions of $45.00 and 0.92 pounds, respectively. The equivalent value of 0.92 pounds is $60.00 for the engine for the MMRPV mission and $208.84 for the engine for the UPT mission which equates system flyaway cost reductions of $131.00 and $333.00, respectively.

Additional benefits of counter-rotation include the reduction of rotor gyroscopic moment loads imposed on engine bearings and structures, engine mounts, and airframe structure. Counter-rotation of HP and LP shafts causes these moments to directly oppose each other and therefore tend to be self-cancelling:

$$M_{GYRO} = I_p \text{ (HP Shaft) } \omega_{HP}\ \omega_{yaw} + I_p \text{ (LP Shaft) } \omega_{LP}\ \omega_{yaw}$$

since $\omega_{LP}$ is negative with respect to $\omega_{HP}$, these terms subtract to reduce $M_{GYRO}$ for counter-rotation.

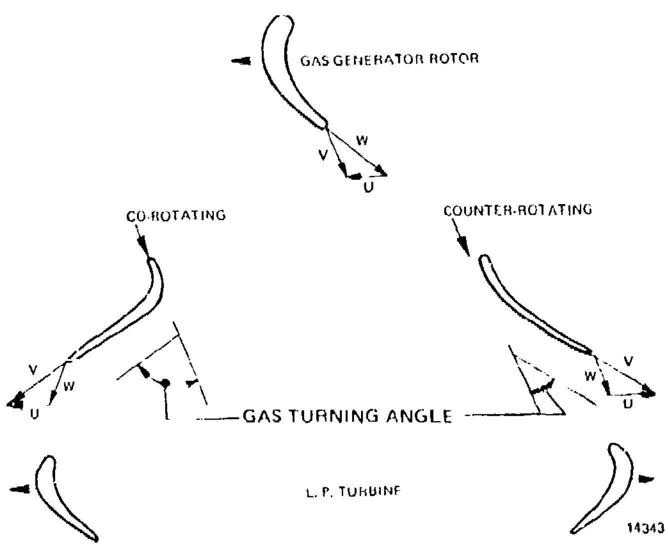



Figure 4.2.10-1. Comparison of Gas Flowpath for Co-Rotating Versus Counter-Rotating Engine HP and LP Shafts.

**CAST LP TURBINE INLET NOZZLE**

| Co-Rotating Shafts | | Counter-Rotating Shafts | |
|---|---|---|---|
| Integral Casting | | Integral Casting | |
| Vanes | = $ 230.00 | Vanes | = $ 210.00 |
| Qualify Casting | = 20.00 | Qualify Casting | = 15.00 |
| Machining | = 350.00 | Machining | = 350.00 |
| Deburring | = 25.00 | Deburring | = 15.00 |
| Insp. | = 25.00 | Insp. | = 15.00 |
| Total | $ 650.00 | Total | $ 605.00 |

Net Cost Reduction = $45.00
Net Weight Reduction = 0.92 pounds

**FABRICATED LP TURBINE INLET NOZZLE**

| Co-Rotating Shafts | | Counter-Rotating Shafts | |
|---|---|---|---|
| Shrouds | = $ 155.10 | Shrouds | = $ 155.10 |
| Cast Vanes | = 160.00 | Strip Stock Vanes | = 20.25 |
| Qualify Castings | = 20.00 | | |
| Machining Shrouds | = 500.00 | Machining Shrouds | = 473.00 |
| Welding | = 60.00 | Welding | = 40.50 |
| Deburring | = 15.00 | Deburring | = 10.00 |
| Insp. | = 25.00 | Insp. | = 20.00 |
| Total | $ 935.10 | Total | $ 718.85 |

Net Cost Reduction = $216.25
Net Weight Reduction = 0.92 pounds

Figure 4.2.10-2. Cost Reduction of LP Turbine Inlet Nozzle Through the Use of Counter-Rotating Shafts.

### 4.2.11 Welded LP Turbine Assembly

The objective of this study was to achieve reduced engine cost by combining the two separate fan turbine rotors into a single welded structure.

In a conventional multiple-stage turbine assembly using integrally bladed cast construction, most of the machining operations are related to the attachment of the individual stages to the rotor assembly.

The two most common techniques employ either a clamped spline connection with outboard diametral pilots, or a multiple dowel bolted attachment.

The welded construction shown in Figure 4.2.11-1 would eliminate virtually all the machining operations associated with conventional attachment designs. The only machining required on the first stage rotor casting shown would be the cut-off and the diametral pilot that forms the weld connection.

The preliminary cost evaluation affirmed the savings that would be realized by eliminating the rotor stage attachment machining. The shaped spline and two pilot diameters were estimated to require 25 hours of machining time. The cost of the automatic weld and the weld machining was estimated to result in a net savings of approximately 1.5 hours in the cost of the rotor stage.

The welded rotor construction necessitates splitting the interstage stator. The most economical construction for the split stator would be a two-piece casting dowelled together prior to final machining of the attachment flange and the interstage labyrinth sealing diameter.

The additional cost associated with the split stator would result from:

1. Two castings instead of a one-piece full round casting.
2. Precision machining of the four interfaces.
3. Drill and reaming for six parting line dowels.

These added stator operations, due to the split construction, were judged to be equivalent to approximately one hour of machining time, thereby reducing the indicated savings to only about 0.5 hours.

The welded construction would also have an adverse effect on the maintenance and replacement cost of a single turbine stage.

In view of the minor initial cost savings indicated for the welded rotor concept, and the adverse factors associated with maintenance and replacement costs, investigation of this cost reduction topic was discontinued.

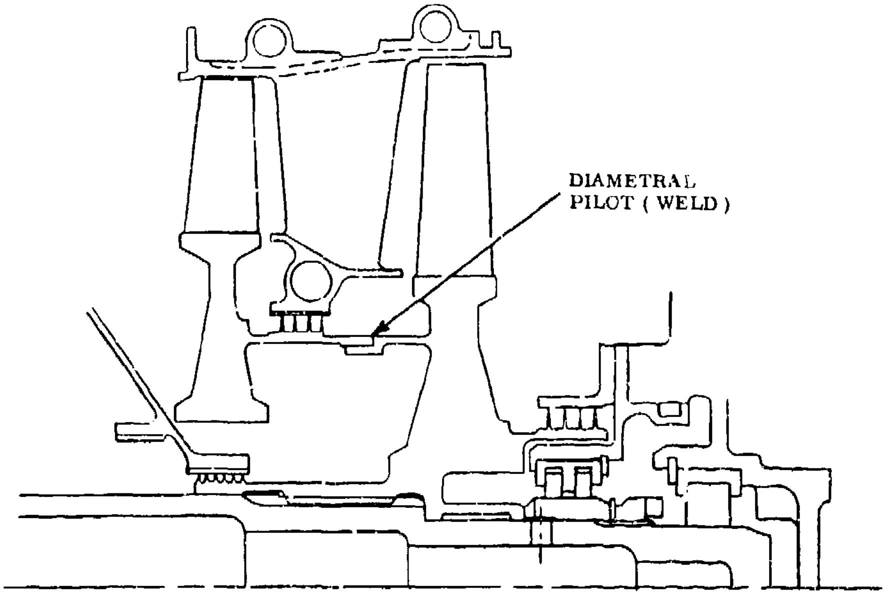



Figure 4.2.11-1. Welded LP Turbine Assembly.

4.2.12 Jet Flap Turbine Blading

A jet-flap principle has been applied to the gas generator turbine to minimize the number of rotor blades. The objective of this study was to determine the feasibility of extending this aerodynamic innovation to the fan turbines to reduce the total number of gasfoils and thereby reduce costs.

A substantial amount of rig test data has been accumulated at Teledyne CAE and by NASA on the jet-flap turbine blade row, jet-flap cascades and complete turbine stages. These results permit an estimate of the potential benefit of the jet-flap fan turbine blading through reduction of the number of blades required for optimized turbine work output, efficiency, and cost reduction.

A Teledyne CAE low pressure turbine design was used to provide design background for this study.

The rotor design was accomplished by prescribing constant total pressure distribution at the rotor exit. Meridional velocity distributions at the rotor inlet and exit were set at nearly uniform values. The exit total temperature was established as a result of the prescribed radial work distribution. Test data from Teledyne CAE and recent NASA programs illustrate the effects of using a "jet-flap" slot to increase the aerodynamic loading capacity of the hub section.

The slot geometry, illustrated in Figures 4.2.12-1 and 4.2.12-2, is based on test data. The slot is proportioned to eject the cooling air flow at critical velocity and normal to the mainstream flow in the region from the hub to one-third blade span.

Design estimates indicate a potential of 1.2 percent efficiency gain or an increased loading of up to 10 percent is available through use of the jet-flap as shown in Figure 4.2.12-3. On the Teledyne CAE turbine design, this allows a reduction of 10 percent in the number of blades without performance penalty.

The airflow required for the jet-flap blade is less than 4 percent of the core gas generator airflow, but this airflow is essentially lost to the engine cycle. A performance-loss equivalent to approximately 1.2 percent SFC results. When combined with the increased costs for fabricating blades with air passages and the jet-flap slot, the cost advantage of the reduction in blade number is far outweighed by uncooled blades.

For growth engines, the low pressure turbine may require cooling because of increased turbine inlet temperature. At such time, the jet-flap slot in the blade will offer significant cost reduction.

The application of the jet-flap technique to low pressure turbine blades will be deferred until such time as increased turbine inlet temperatures dictate a requirement for cooling these stages.

4.2.13 Inter-Shaft Sealing

The objective of this study was twofold: first, to reduce engine cost and weight, and second, to improve engine reliability. This was envisioned as a possibility by replacing the two inboard face seals in the front bearing cavity with a non-contacting, positive, inter-shaft seal. In conventional two-shaft engine constructions, the cavity between the shafts is generally vented to approximately compressor discharge static pressure. Venting avoids the accumulation of liquid in the high pressure spool and also acts as a buffer against leakage of hot gas and/or oil in the intershaft cavity.

The high pressure gas in the intershaft cavity is usually sealed by two face-type contact seals, each sealing between a shaft and the stationary structure.

This cost reduction topic proposes the use of an oil dam, formed between the two shafts, to seal the intershaft gas pressure and to replace the two conventional face-type contact seals.

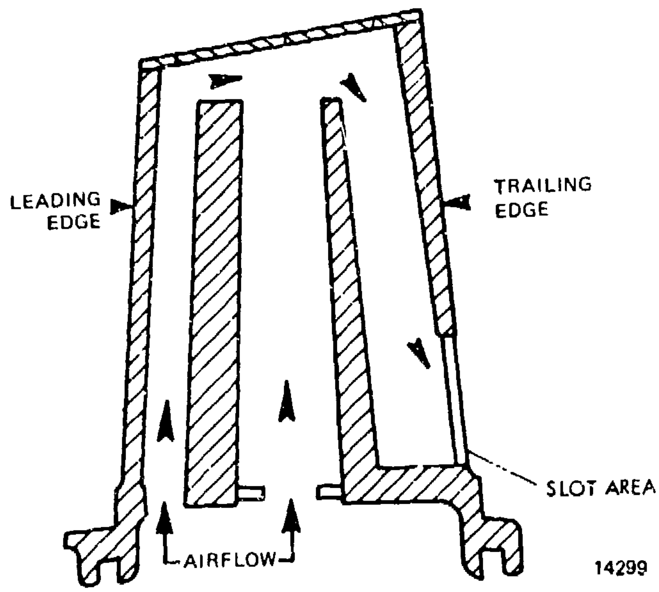



Figure 4.2.12-1. Typical Cooled Turbine Blade Schematic Illustrating Jet Flap Airflow.

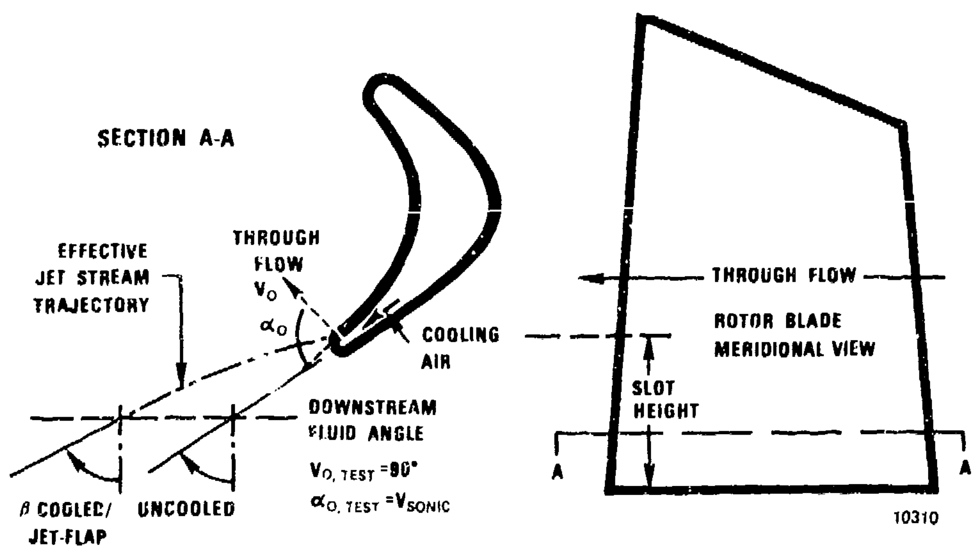



Figure 4.2.12-2. Typical Cooled Turbine Blade Illustrating Jet Flap Aerodynamic Effects.

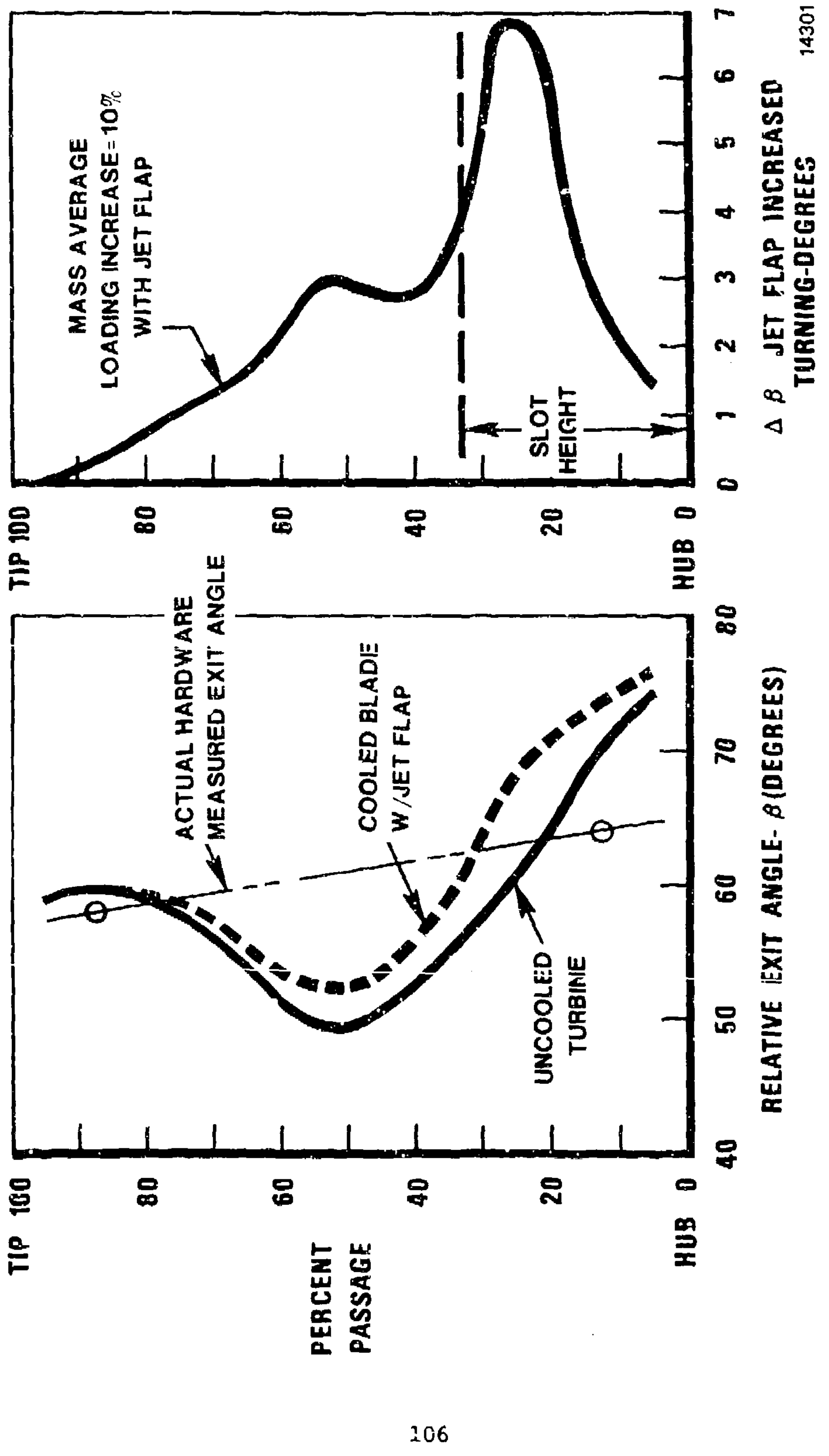



Figure 4.2.12-3. Jet Flap Effects on High-Work Turbine.

The preliminary design of the oil-dammed gas seal is shown in Figure 4.2.13-1. A continuous supply of oil is directed into a rotating cavity formed in the high pressure shaft assembly. A disc on the fan shaft assembly separates the oil into two columns. Any external pressure differential results in a height shift of the two columns, analogous to the familiar manometer operation.

Calculations indicate that the liquid dam will support the expected peak intershaft cavity pressure differential of 140 psi.

A 0.71 HP loss is calculated for the liquid dam gas seal.

An approximate weight comparison was made between the liquid dam gas seal concept and the conventional two-face seal design. The dam seal was found to be approximately 0.7 pounds lighter due primarily to the actual weight savings at the two face seals and the stationary structure required to support the face seal housings.

The hardware required to implement the liquid dam seal are basically simple metal components amenable to low cost production techniques such as precision casting and powdered metal fabrication. The four details required for the liquid dam seal have been estimated to cost about $6.00 total, in production. The two face-type contact seals generally cost about $35.00 each in production. The precision machined shaft runner that provides the contacting surface for the face seal generally costs about $15.00 each in production. Additional savings are realized by the elimination of several stationary structure machining operations. The overall savings afforded by the liquid dam gas seal concept is approximately $100 per engine. The addition of G & A and Profit and application of the weighting factors for engine weight related to mission produces the following system flyaway cost reductions:

| | | |
|---|---|---|
| UPT (two engines) | = | $434.00 |
| MMRPV | = | $190.00 |

An additional beneficial feature of the liquid dam seal is its basic simplicity and non-contacting mode of operation. In contrast, face-type

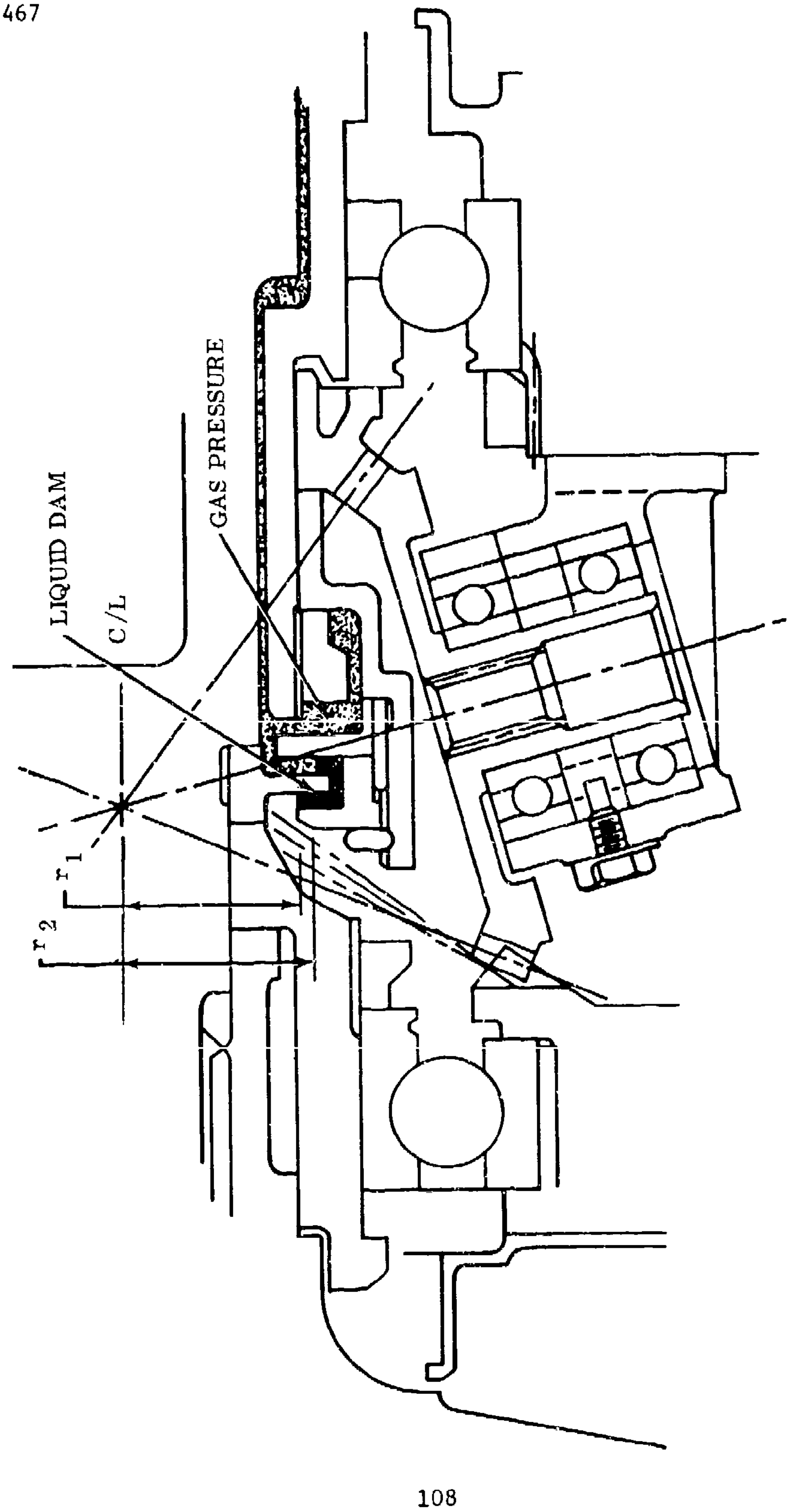



Figure 4.2.13-1. Preliminary Design of Oil-Dammed Gas Seal.

contact seals are sensitive to both environmental and dimensional conditions. With normal operating circumstances, face seals are subject to wearout rates which involve both reliability and maintainability costs.

The small cost reduction potential versus a substantial development cost (seal rig design, fabrication and testing) make the net payoff of this item small. Additionally, the calculated average rotational speed of the oil is reduced by approximately 70 percent for counter-rotating shafts, thus reducing the pressure differential which can be supported well below an acceptable level.

### 4.2.14 Gas Foil Bearings

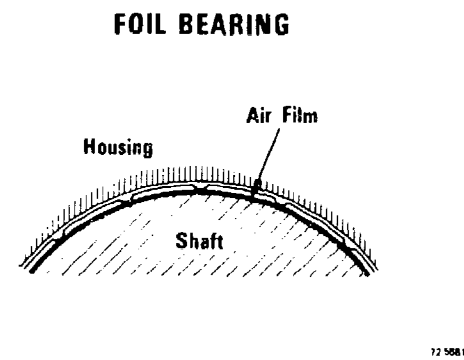
The objective of this study was to achieve cost and weight reductions through the use of gas foil bearings to replace rolling element bearings and related lubrication system components.

Gas foil rotor bearings can provide considerable cost reduction by eliminating the normal circulating lube system and its attendant oil pump, lubrication lines, fittings, galleries and scavenge system.

The low cost concept considered is the airfoil bearing configuration shown in Figure 4.2.14-1. This bearing uses the working medium of the engine for its lubricant and essentially introduces no loss of engine cycle heat. The bearing is comprized of a thin tape having a series of raised, resilient, formed supports. The supports provide radial compliance and Coulomb and viscous damping. Consequently, the bearing is stable for high speed rotating systems and provides effective damping for control of shaft vibration. The thickness of the hydrodynamic air wedge, which separates the journal from the bearing, is dependent on the viscosity of the air as well as its density, and the compressor discharge air provides an ideal environment for successful bearing performance.

To determine the benefits of using a gas foil bearing, an analytical study was performed on the baseline engine which uses an oil-lubricated roller bearing located downstream of the turbine. Replacement of this bearing with an air bearing would accomplish the following:

Cost would be reduced through the elimination of parts.



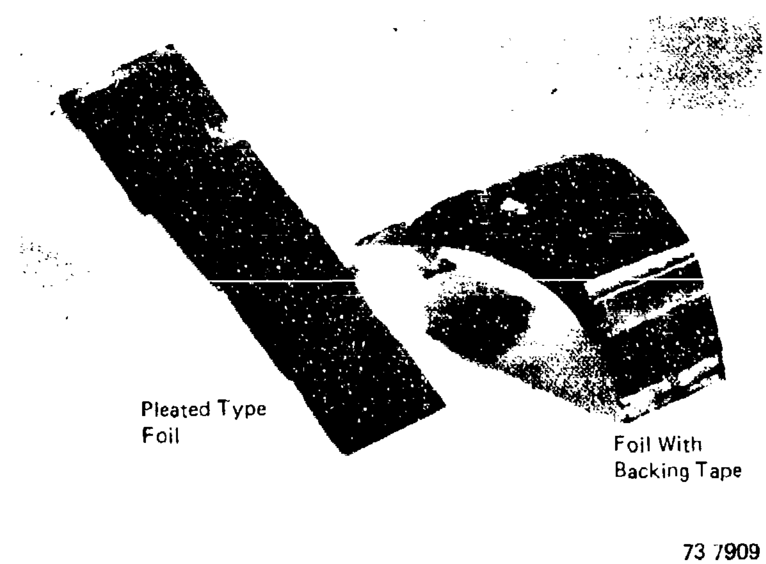




Figure 4.2.14-1. Gas Foil Bearings.

The rear bearing lubrication system would be eliminated, along with lubrication lines, fittings, galleries, and scavenge pump.

The oil cooler size and weight would be reduced.

Engine weight would be reduced an estimated one percent.

The rear bearing could be moved upstream of the turbine rotor, allowing the bearing support to be carried through the turbine nozzle. The jet exhaust and nozzle would, therefore, become non-load-bearing, simplifying the rear end design and reducing cost.

Cooling of the bearing cavity would no longer be required, permitting the option of raising the turbine inlet temperature without the consideration of the affects on the rear bearing.

The potential benefits listed above precipitated the analysis encompassing shaft critical speeds and bearing loads. The first step in the analysis was to determine the rotor characteristics of the HP shaft which are as follows:

| | |
|---|---|
| Shaft Weight - lbs | 81.21 |
| Bearing Span - inches | 11.15 |
| Shaft Mass Polar Moment of Inertia - lb-in-sec$^2$ | 1.424 |
| Shaft Mass Transverse Moment of Inertia about CG - lb-in-sec$^2$ | 4.94 |
| Shaft CG (from centerline of front bearing) - inches | 8.43 |
| Maximum Rotor Speed - percent | 100 |

The calculated critical speeds of the HP rotor-bearing system are listed below for the range of bearing stiffnesses.

| Bearing Stiffness (lbs/in) | First Critical Speed (%) | Second Critical Speed (%) | Third Critical Speed (%) |
|---|---|---|---|
| $10^4$ | 8.5 | 14.3 | 144 |
| $10^5$ | 23 | 46 | 153 |
| $10^6$ | 51 | 130 | 223 |

The stiffness of the air-lubricated foil bearings will depend on the size and geometry selected. For 5-inch diameter journal bearings, this stiffness will be in the range of $10^5$ to $10^6$ lb/in. Thus, the maximum operating speed of the HP rotor is well below the third critical speed of the rotor-bearings system. However, the above data also shows that over the range of bearing stiffnesses, $10^5$ to $10^6$ lb/in, the second critical speed may be below, coincident with, or above the second critical speed of the rotor/bearings system. In the actual application, the bearing stiffness would have to be adjusted to maintain an adequate margin between the operating speed and the critical speed. This can be done by proper selection of foil compliance, bearing clearance, and pre-load.

With the critical speed versus bearing stiffness showing the capability of operating within the shaft and bearing requirements, the next step was to determine the bearing loads. Figure 4.2.14-2 shows the calculated journal bearing loads for level flight, landing, and 3.5 radians/second yaw maneuver. The present design allows for 5-inch diameter journal bearings. Thus, the start-up loads are very small, about 2 psi, assuming a length-to-diameter ratio of unity. The most critical condition is that associated with the 3.5 radians/second yaw maneuver. Under this condition, the maximum bearing load would approach 72 psi which is four or five times the unit loading demonstrated to date on foil bearings. Although this is only a short duration maneuver load, it indicates that an increase in bearing span to reduce the bearing reactions to the gyroscopic moment induced in the maneuver, as well as an increase in bearing size need to be considered, with the objective being a reduction in the maximum radial unit loading. The present capacity of journal foil bearings has been demonstrated to 20 psi unit loading with expected increases up to 30 to 35 psi available in the near future. An increase in bearing span will have a linear effect on the reduction of the bearing loads. Thus, twice the bearing span would halve the gyroscopic moment loadings on the bearings. It can be seen that bearing span dimensions can only be increased minimally before the engine becomes longer and heavier, cancelling the potential advantage of using the gas foil bearings.

An increase in bearing size can be utilized to reduce the radial unit loading. An increase in bearing diameter can only be accommodated to about a 6-inch diameter journal before it would begin to affect the aerodynamic flowpath of the engine. An increase in bearing length beyond a length-to-diameter ratio of unity at the diameters (which is being investigated) begins to be self-defeating because the alignment capability to provide film convergence over the full leaf begins to drop off, lowering the unit load capacity.

Although the gas lubricated foil bearings display potential cost and weight savings, it is recommended that further investigations be suspended for the baseline engine until such time that the unit loading capacity of these bearings is considerably enhanced.

### 4.2.15 Simplified Lubrication System

The objective of this study was to achieve cost and weight reductions by means of a simplified lubrication system. To determine the best possible lubrication system, a number of candidate systems were investigated for feasibility and a matrix of the combinations (candidate systems versus area lubricated) was made.

| | Front Bearing Load (lb) | Rear Bearing Load (lb) |
|---|---|---|
| Level Flight | | |
| Non-Rotating Load Component | 20 | 61 |
| Rotating Load Component | 100 | 100 |
| Landing | | |
| Non-Rotating Load Component: | | |
| 10G Down = 812.1 lb | 198 | 614 |
| 2G Side = 162.4 lb | 39 | 123 |
| ± 14 Rad/sec$^2$ Pitch Accel. = 69.16 in/lb | 6 | 6 |
| ± 6 Rad/sec$^2$ Yaw Acceleration = 29.64 in/lb | 3 | 3 |
| Vertical | 204 | 620 |
| Horizontal | 42 | 126 |
| Combined | 208 | 632 |
| Rotating Load Component: | 100 | 100 |
| 3.5 Radians/Second Yaw Maneuver | | |
| Non-Rotating Load Component: | | |
| Gyro: 3.5 Rad/sec in Yaw = 18,266 in/lb | 1638 | 1638 |
| ± 1G Vertical = 81.21 lb | 20 | 61 |
| Combined | 1658 | 1699 |
| Rotating Load Component: | 100 | 100 |

Figure 4.2.14-2. Summary of Journal Bearing Loads for HP Shaft of Baseline Engine.

The candidate systems are:

1. Full Lube (Recirculating Oil)
2. Grease Packed
3. Expendable Oil Mist - Figure 4.2.15-1
4. Recirculating Oil Mist
5. Fuel Lube (Recirculating Fuel)
6. Wet Oil Sump - Figure 4.2.15-2
7. Oil Wick -- Figure 4.2.15-3
8. Dry Lube
9. Full Lube (Replace Scavenge Pumps)

The areas lubricated are:

1. Front Bearings (Ball Type)
2. Rear Bearings (Roller Type)
3. Accessory Drive (Gears, Bearings and Splines)

The matrix of the above combinations is shown below with an F indicating a feasible system and an N indicating a non-feasible system.

| | Recirc. Full Lube | Grease Packed | Expendable Oil Mist | Recirc. Oil Mist | Fuel Lube | Wet Oil Sump | Oil Wick | Dry Lube | Full Lube |
|---|---|---|---|---|---|---|---|---|---|
| | 1 | 2 | 3 | 4 | 5 | 6 | 7 | 8 | 9 |
| Front Bearings | F | N | N | N | F | F | F | N | F |
| Rear Bearings | F | N | F | F | N | N | N | N | F |
| Accessory Drive | F | F | F | F | F | F | N | F | F |

From the above matrix, it can be seen that only two systems (both full lube) can accommodate all areas, but many hybrid systems are feasible and could be combined with the other systems. The feasibility of the systems in the matrix were determined without regard to any interconnection between the areas. For example, a fuel lube system may not be practical because it could contaminate another system.

Starting the final evaluation with the rear bearings requirements, the matrix showed that only four feasible systems are presently available. The expendable and recirculating oil mist systems were eliminated to ensure compliance with the life and reliability requirements of a highly loaded high "DN" bearing. In addition, these systems have the disadvantage of requiring a volume increase because of the requirement for an oil reservoir. In the case of the expendable oil system, the oil mist from the HP turbine bearing would be dumped into the LP turbine flowpath. With these systems eliminated, only the full lube systems remain acceptable.

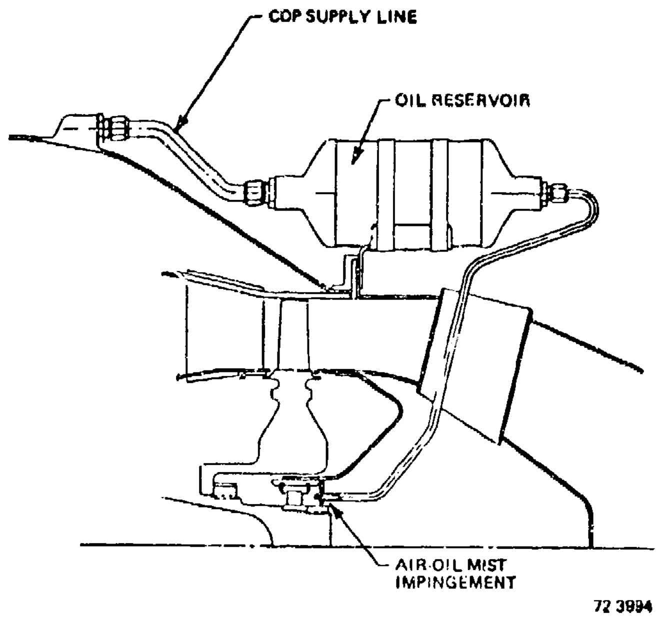



Figure 4.2.15-1. Typical Teledyne CAE Expendable Oil Mist Rear Bearing Lube System.

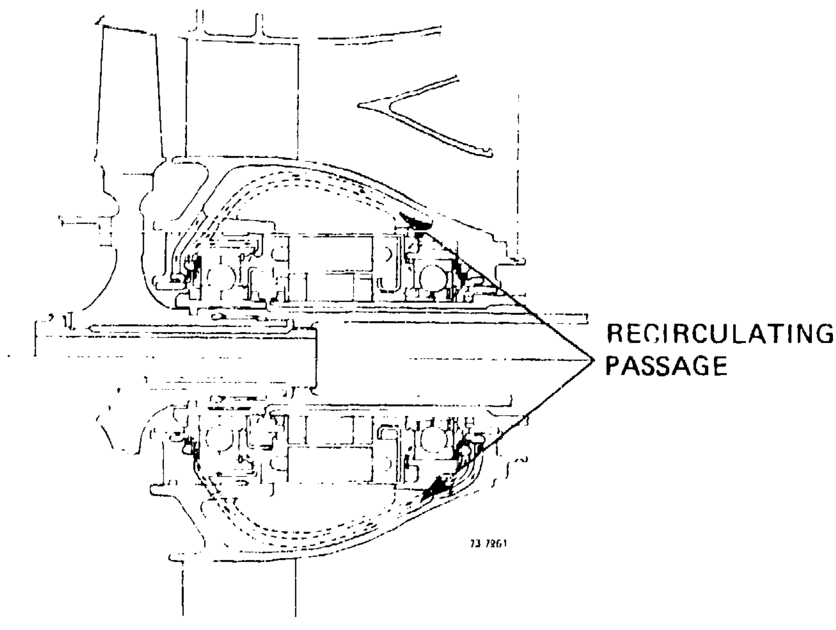



Figure 4.2.15-2. Typical Teledyne CAE Front Bearing Wet Oil Sump Lube System.

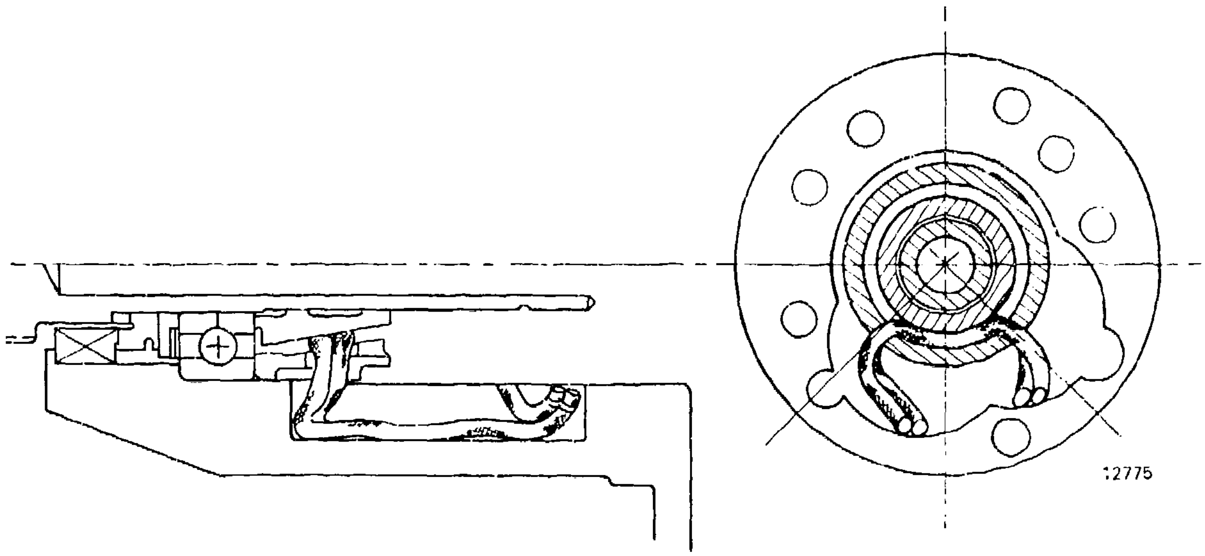



Figure 4.2.15-3. Typical Oil Wick Lubrication System.

A full lube system requires an oil reservoir, pressure pump and scavenge elements with drive and related valving.

The front bearing cavity had five lube system candidates: oil wick, wet sump, fuel lube and both of the full lube systems. The oil wick system was eliminated after heat generation numbers were calculated and considered too high for this system. The wet sump system (in the present state-of-the-art) has several problems which eliminated it from further consideration. Extreme attitudes require pumps to ensure lubrication of both the HP and LP bearings with pressurized oil flow to adequately lubricate the high DN value bearings. Since a tower shaft is required for the accessory drives, it would be necessary to provide seals to keep the oil from draining through the tower shaft strut and into the accessory case, and th address long-term storage requirements. A fuel lube system for the rear bearing cavity would require another complement of pumps, valves, etc., to be added to the existing oil system to recirculate the fuel.

After evaluating all candidate systems, the full lube systems presently remain the only acceptable solutions for the front bearing compartment. The use of this system for the front compartment adds very little to the complexity of the engine since the rear bearing lube system has all the required components except for the scavenge elements. The accessory drive gears and bearings could be lubricated by many systems, but will have oil draining from the front bearing compartment, through the tower shaft strut, into the accessory case. The oil must then be returned to the oil tank.

Therefore, provision for scavenging the oil in the accessory case sump becomes the only additional requirement to the system.

A cost and weight comparison was made of the remaining two circulating lubrication systems. The conventional full lube circulating lubricating system has a pump with one pressure element and four scavenge elements. The alternate full lube circulating lubrication system has high-speed rotating pumping elements in each bearing cavity mounted directly on the LP and HP shafts. A typical pumping element is shown in Figure 4.2.15-4. These screw-thread-type pumps replace the four scavenge elements that are necessary in the conventional system. Cost and weight comparisons are presented in Figure 4.2.15-5 with the X symbol denoting common usage in both systems.

Using the weighting factors for the respective missions, the system fly-away cost reductions are:

| | | | |
|---|---|---|---|
| UPT | (two engines) | - | \$712 |
| MMRPV | | - | \$296 |

These dollar values do not take into account the risk involved in developing this alternate system. The reliability and maintainability of both lubrication systems appear to be relatively the same with the life cycle cost advantage going to the lower cost system. The capability of this type of pumping element to provide sufficient head for operating at extreme attitudes would require engineering development. During the development program, the capacity of the pumping elements would be determined and an assessment made of the reliability of the combined lubrication and bearing systems.

The alternate system looks attractive for RPV and fighter applications with extended level-flight operation. For the UPT application, greater capability will be required of the pumping system through all attitudes. For this reason, it is recommended that the full circulating system be used at this time.

### 4.2.16 High Speed Accessory Drive

The objective of this study was to show improved engine performance through the use of "thin struts" in the primary flow transition duct to obtain the same performance with a smaller engine (lighter and less cost). The baseline engine has been designed to incorporate a high-speed tower shaft with struts sized accordingly. The thin struts in the primary transition duct improve the duct pressure recovery characteristics relative to the baseline system that uses thick structural struts and tower shaft struts. The size of the struts is directly dependent on the size of the tower shaft. The tower shaft size and speed are dictated by torque and critical speed requirements.

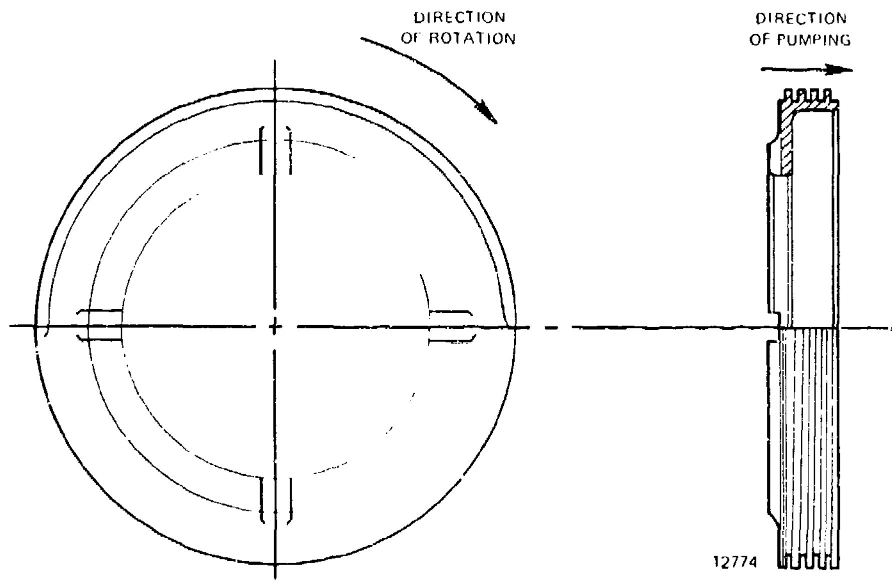



Figure 4.2.15-4. Typical Teledyne CAE High-Speed Rotating Pump Element.

Standard design practice uses a relatively low speed shaft to satisfy the critical speed requirements which results in a large diameter shaft to satisfy the increased torque requirements. This, in turn, requires a thick strut with its attendant high pressure recovery loss.

For transmission of a given horsepower, the torque is inversely proportional to the speed. Changing the shaft speed from 10,000 rpm to 50,000 rpm lowers the torque requirement to one-fifth of the original value. The torsional stress is proportional to the torque and inversely proportional to the radius to the third power. This allows the high speed shaft to be sized at less than 60 percent of the diameter of the low speed shaft while maintaining the same stress levels.

Using the high speed small diameter shaft, the problem is to satisfy the critical speed requirements. One solution is to use a Be/Ti composite shaft which has been presented as a candidate cost reduction topic. Another solution to meet critical speed requirements would be the use of a mid-span bumper bearing to alter the shaft critical speed.

Using the cost and weight studies performed on the baseline engine and scaled versions of half-thrust and twice-thrust, the curves of relative cost versus thrust, and relative weight versus thrust are shown in Figure 4.2.16.

| CONVENTIONAL VERSUS ALTERNATE FULL CIRCULATING LUBRICATION SYSTEMS | Cost($) | Weight (lbs) | Cost ($) | Weight (lbs) |
|---|---|---|---|---|
| Pump with 1 pressure and 4 scavenge elements plus filter | 377.50 | 5.45 | --- | --- |
| Pump with 1 pressure element plus filter | --- | --- | 75.00 | 1.36 |
| Screw thread pumping elements (5 pairs) | --- | --- | 137.50 | 2.53 |
| Lubrication lines | Y | X | Y + 34.00 | X |
| Oil Tank | X | X | X | X |
| Heat Exchanger | X | X | X | X |
| Increment charged to the system | 377.00 | 5.45 | 246.50 | 3.89 |
| Savings | --- | --- | 130.50 | 1.56 |

Figure 4.2.15-5. Lubrication Systems Cost and Weight Summary.

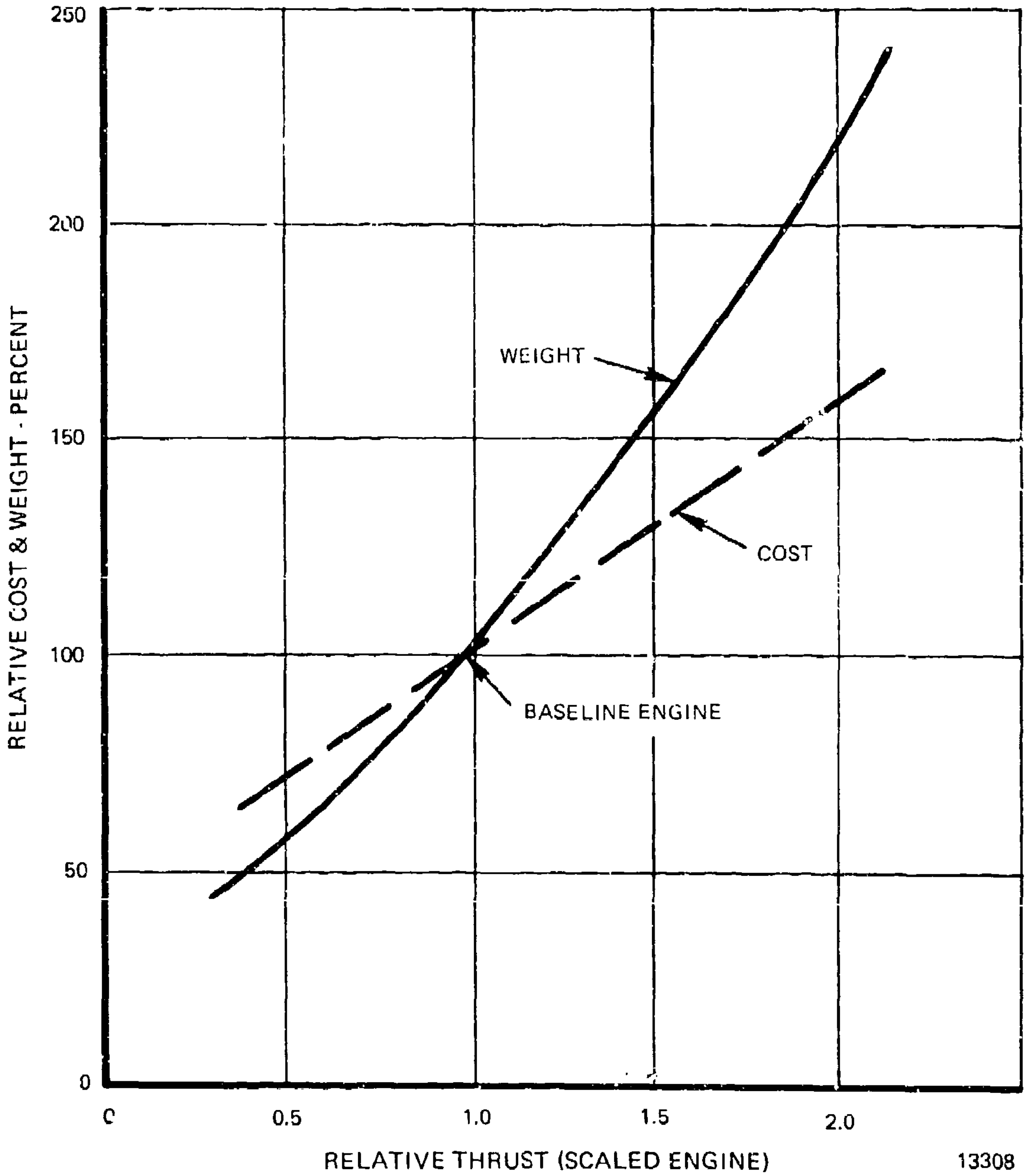



Figure 4.2.16. Relative Cost and Relative Weight Versus Thrust.

The use of thin struts improves the performance, thereby allowing the pressure recovery duct to be resized with an attendant change in the weight of the engine. Using the curves of Figure 4.2.16, it can be seen that as engine weight is reduced, the cost is reduced. Actual values for cost and weight savings require detailed analyses of the aerodynamics, engine performance and mechanical configuration.

Although it is apparent that a cost and weight advantage can be derived from the use of a high speed accessory drive, further studies to provide actual values are not recommended until specific engines are defined for detailed analysis and cost/weight tradeoffs.

### 4.2.17 Controls and Accessories

The objective of this study was to examine the cost and weight reduction potential of three currently available advanced technology components, and compare them to similar currently used production components. The three components evaluated were: (1) electronic fuel control logic system, in combination with a rotary metering valve: (2) a low-cost, variable displacement piston pump; and (3) a fluidic logic system and air motor actuators to control inlet guide vane and interstage bleed valve position.

The baseline engine for these cost analysis studies has functional control system requirements which can be related to the current J100-CA-100 engine requirements. The approach used in making the cost analysis was to summarize the required functions and compare them to the functions of the current J100 control system, then add those control capabilities required to enable the current J100 system to control the baseline engine shown in Figure 4.2.17-1.

As indicated in Figure 4.2.17-1, the current technology system, as adapted through additional functions to properly control the turbofan engine, is costly and heavy. The advanced electronic control system cost is approximately 40 percent of that of the current system, weighs 26.15 pounds less (which translates to an additional system cost saving of $11,088/vehicle, based on value of engine weight reduction for a trainer application) and has a much smaller volume.

#### Electronic Control

The electronic fuel control system consists of three subsystems: the engine governor and control subsystem (electronic module); the start and supervisory subsystem (electronic module); and the fuel metering subsystem (electronic motor and hydromechanical valve). The two electronic modules, chassis-mounted, are presently in an industrial configuration and can be readily repackaged for flight. The fuel metering subsystem consists of a stepper motor and a fuel metering valve. Since the three subsystems are modular and small, they can be located in a variety of places - this allows great flexibility in controls/engine and engine/airframe interfacing.

| | | | | J100-CA-100 | | | | Addition of Capability to J100 System | | |
|---|---|---|---|---|---|---|---|---|---|---|
| Function | Requirement | Cost ($) | Weight (lbs) | Function | Requirement | Cost ($) | Weight (lbs) | Function | Cost ($) | Weight (lbs) |
| Fuel Flow<br>Isoch. Governing<br>$T_2$ Compensation<br>$P_2$ Compensation<br>$P_3/P_2$ Fuel Schedule<br><br>Var. Geo. Schedule<br>LP Shaft Speed Limit<br><br>$W_F$/CDP Accel.<br><br>Fuel Pump Integral | 2000 pph | 3,800 | 11.0 | Fuel Flow<br>Isoch. Governing<br>$T_2$ Compensation<br><br>Bleed Valve Schedule &<br>Actuator<br>$W_F$/CDP Accel.<br><br>Fuel Pump | 2000 pph | 8,000 | 25 | $P_2$ Compensation<br>$P_3/P_2$ Fuel Schedule<br>Turbine Temp. Limit<br>Var. Geo. Schedule<br>LP Shaft Speed Limit | 50<br>150<br>2,000<br>750<br>250 | 0.25<br>2.0<br>6.0<br>2.0<br>1.5 |
| Turbine Temp. Limiting<br>Bleed Valve<br>Var. Geo. Actuators | | 200<br>395<br>742 | 1.0<br>2.5<br>1.5 | Bleed Valve | | 385 | 3.2 | Var. Geo. Actuators | 1,080 | 2.2 |
| SYSTEM TOTALS | | 5,137 | 16.0 | | | 8,385 | 28.2 | | 4,280 | 13.95 |
| | | | | COMBINED CURRENT SYSTEMS | | 12,665 | 42.15 | | | |
| NET COST REDUCTION | | 7,528 | | | | | | | | |
| NET WEIGHT REDUCTION | | | 26.15 | | | | | | | |

Figure 4.2.17-1. Baseline Engine Control System Requirements Compared to J100 Control System.

Rotor Metering Valve

To be functional and cost effective on a wide range of operational engines, a fuel metering unit must have the following characteristics:

Accurate (minimum metered flow of 20 to 70 pph with 3 to 5 pph accuracy)

Low Cost

Reliable (especially clog-free)

Positive Actuation

Fail-Fixed Capability

After reviewing and rating a number of metering techniques, Teledyne CAE determined that the rotary valve/stepper motor approach is by far the most promising. Teledyne CAE has experience with all of the techniques investigated.

The two critical characteristics in which the rotary valve/stepper motor excels are accuracy and fail-fixed capability. Small gas turbine engines require highly accurate metered fuel flow at various engine operating conditions. The most stringent accuracy is required when the control is on starting or deceleration mode at low flows (20 pph), with a possible accuracy requirement of 3 pph. The rotary valve satisfies this requirement.

The rotary valve/stepper motor fail-fixed feature provides fixed flow when the flow package loses electrical power. A permanent magnet stepper motor, geared to the metering valve, holds the valve in its last commanded position. For manned aircraft, executive action can then be taken to override the control.

Fuel Pump

Although a variety of candidate pumping systems can be used, centrifugal, piston, gear, or various combinations depending on the specific requirements, a variable displacement piston pump developed by the National Gas Turbine Establishment (NGTE) appears to have great potential as a central element in a low-cost simplified control system.

The NGTE pump (Figure 4.2.17-2) consists of three axially-disposed plungers housed within a rotor. Plunger stroke is determined by a variable-angle cam plate controlled by a servo piston. Operating pressure is regulated by a spill valve that maintains a constant delivery pressure.

This pump has the capability of providing a simple governor, fuel metering, and a pumping system at under $1,000 production cost and 4 to 5 pounds in weight.

Fluidic Logic for IGV, IBV Control

Variable inlet guide vanes (IGV's) and interstage bleed valves (IBV's) prevent starting problems in high-pressure-ratio axial compressors. The variable IGV lowers the compressor blade angle-of-attack at low rpm's and thus reduces blade loading. The IBV lowers blade loading by reducing back pressure on the compressor's front stages.

IGV's and IBV's must be actuated by services that are in turn driven by logic systems. Teledyne CAE has reviewed one promising logic device and two actuation devices.

Fluidic Logic System

The Bendix fluidic logic system schedules vane (or bleed valve) position as a function of compressor corrected speed. The logic compares pressures at four locations (interstage, compressor discharge, compressor inlet and engine compartment or fan discharge) - and then commands the actuator.

Fluidic Actuator

The Bendix actuator (Figure 4.2.17-3) operates by pressure differentials across a diaphragm. The advantage of this actuator is that the logic and actuator can be packaged in a compact unit and bolted directly to the engine (in the case of the IBV, the logic, actuator, and bleed valve can be packaged in one unit and bolted directly to the engine bleed port).

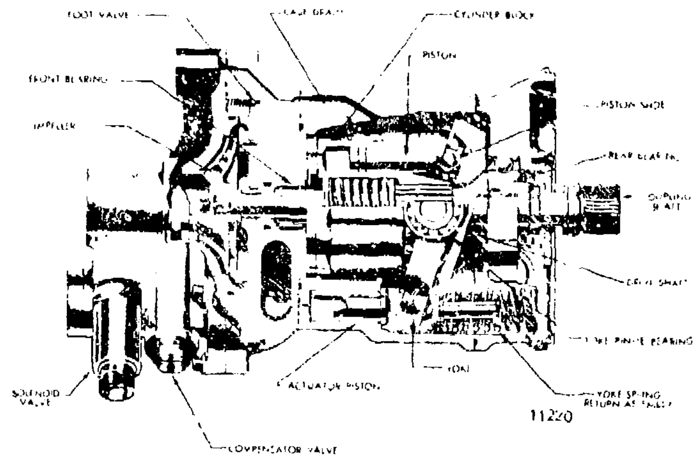



Figure 4.2.17-2. Typical Swash Plate Fuel Pump Offers Promise as a Central Element in a Simple, Low-Cost Control Scheme.

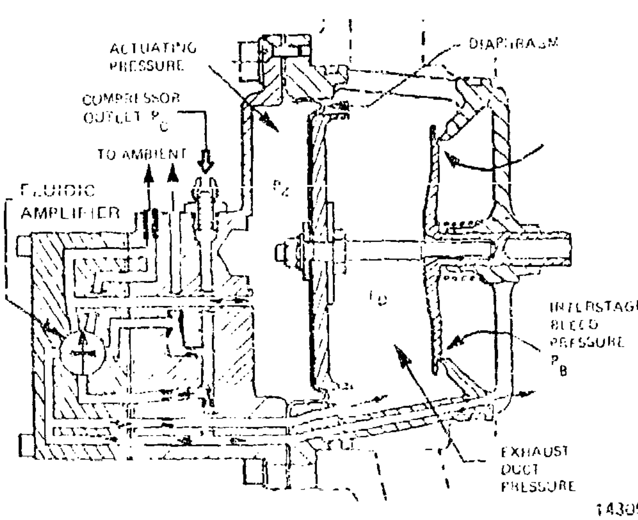



Figure 4.2.17-3. Bendix Fluidic Actuator.

Air Motor Actuator

The CECO air motor is compact and low cost. The motor is simply a shaft driven at high speed by a small air turbine. The turbine is driven by compressor bleed air after engine start and by bottled compressir air during engine start.

The Bendix System (fluidic logic plus fluidic actuator) is already being evaluated on another Teledyne CAE program for use in driving variable IGV's.

Self Test and Diagnostics

The ability to detect a control system malfunction within a hydromechanical system is limited or impractical, difficult, and expensive. The advanced electronic unit will have a malfunction detection capability built into the soft-ware. For RPV applications, this self-test feature will, on detection of a malfunction, switch the control to "fail-fixed" mode to assure mission completion. In man-rated applications, this feature will also signal the pilot that a control system failure has occurred, thus enabling executive action to be taken while the control remains in the fail-fixed mode until override.

The electronic control is readily adaptable to engine diagnostic and condition monitoring systems. By means of an on-board events recorded, engine control and other critical parameters can be monitored and recorded for analysis by ground support equipment. Cost of this on-board equipment is on the order of \$6-20,000 per aircraft, depending upon the degree of sophistication employed, but should return many times that in preventative maintenance.

Also under investigation at Teledyne CAE is a method of locating degradation through spectral analysis. The engine noise characteristics are recorded spectrographically at intervals, and variations can be attributed to degradation of various components by signature analysis.

In view of the \$13,000 minimum savings available per engine, it is recommended that the necessary programs to develop these control system techniques be pursued.

Teledyne CAE further recommends that on-board diagnostics and signature analysis be considered for future integrated instrumentation/diagnostic display systems.

4.2.18 Engine Specification Changes

The objective of this task was to review military specifications applicable to engine design, development and testing, identify inappropriate requirements, and recommend cost reducing changes.

MIL-E-5007D* was selected as the prime source of general technical requirements for engines. MIL-E-5007D and its U.S. Navy counterpart (AS2684) are intended to provide guidance (see Section 2.2.1) to technical specialists of DOD procuring activities in imposing requirements on specific engines.

In Teledyne CAE's previous experience, the requirements of MIL-E-5007D have been considerably modified through coordination between Teledyne CAE and procuring activity specialists to ensure the imposition of cost-effective requirements. Such modifications are included in the model specification for the J69-T-25 (T-37 trainer) and the J69/J100 (RPV series).

This experience was applied to assess the requirements of MIL-E-5007D as they apply to a UPT or MMRPV engine. Figure 4.2.18-1 provides a tabulation of 5007D requirements (by paragraph number and title). Adjacent to each requirement is listed the status of that requirement in a currently flying engine together with a comment on its probable need in the UPT or MMRPV missions.

During the APSI studies, the Aerospace Industries Association (A.I.A.) concurrently completed a review of MIL-E-5007D which re-emphasized the need to critique that specification at each instance of specific application. The A.I.A. study reported[1] (Figure 4.2.18-2) that "blanket" imposition of MIL-E-5007D could have a major development cost impact as shown in Figure 4.2.18-2 (which is reproduced from that report).

In a subsequent definition of derivative engines for specific applications, Teledyne CAE suggests that critiquing of specifications for cost-effectiveness be continued as a formal task. The systems cost methodology/techniques evolving at Teledyne CAE as a result of APSI participation, furthered by the JTDE life cycle cost computer model for engine design, will be of immense benefit in formal evaluation of specification cost effectiveness. Some specific topics for critique are discussed below.

4.2.18.1 Armament Gas Ingestion (3.1.2.10.6)[2]

The need for inclusion of an engine capability to operate with armament gas ingestion obviously depends on the mission. If the USAF and weapons

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* MIL-E-5007D, Military Specification Engines, Aircraft, Turbojet and Turbofan - General Specification for, (15 October 1973)

1 Aerospace Industries Association of America, Inc., Project ACE Recommendations; 1 March 1974

2 Paragraph numbers from MIL-E-5007D are shown in parenthesis.

| SPECIFICATION REQUIREMENT | | ENGINE/MISSION/APPLICABILITY | | | |
|---|---|---|---|---|---|
| NO. MIL-E-5007D PARA. NO. | SECT. 3 REQUIREMENT (SUBJECT) | TRAINER ENGINE J69-T-25 | 455 | MULTI MISSION RPV ENG. J69-T-41A | 455 |
| 3.1.2.10.6 | Armament gas ingestion | No | N/R | No | Opt. |
| 3.1.2.11.3 | Bleed air contamination | No | Yes | No | N/R |
| 3.1.2.12 | Radar cross section | No | N/A | No | Opt. |
| 3.1.2.5.1 | Gyroscopic moments | Yes | Yes | Yes | Yes |
| 3.2.1.5.1 | Operating attitudes & conditions | Yes | Yes | Yes | Yes |
| 3.2.1.5.6 | Thrust transients | Yes | Yes | No | Yes |
| 3.2.1.5.7 | Windmilling | Yes | Yes | No | Opt. |
| 3.2.4.1 | Maintainability numerical requirements | No | N/R | No | N/R |
| 3.2.5.2 | Icing conditions | Yes | Yes | No | Opt. |
| 3.2.5.6.1 | Bird ingestion | No | Opt. | No | Opt. |
| 3.2.5.6.2 | Foreign object damage | No | Yes | No | Opt. |
| 3.2.5.6.3 | Ice ingestion | No | Yes | No | Opt. |
| 3.2.5.6.4 | Sand ingestion | No | Yes | No | N/R |
| 3.2.5.8.1 | Exhaust smoke emission | No | Yes | No | Opt. |
| 3.2.5.8.2 | Invisible exhaust mass emissions | No | Yes | No | N/R |
| 3.3.1.2.6 | Screw threads | No | Opt. | No | Opt. |
| 3.3.8.7 | Design material properties | No | N/R | No | N/R |
| 3.3.8.9.1 | Containment | No | Yes | No | Opt. |
| 3.3.8.9.3 | Disc burst speeds | Yes | Yes | Yes | Yes |
| 3.5.1.2 | Maintenance inspection techniques | No | Yes | No | Yes |
| 3.7.1 | Anti-icing system | No | Opt. | No | Opt. |
| 3.7.3.1.1 | Fuel flowmeter | No | Opt. | No | N/R |
| 3.7.3.3.2 | Fuel contamination | No | Yes | No | Yes |
| 3.7.4.1 | Electrical power | No | Opt. | No | Opt. |
| 3.7.6.6 | Thrust indication | No | Yes | No | Yes |
| 3.7.7.4.1 | Oil reservoir | No | Opt. | No | Opt. |
| 3.7.7.4.3 | Oil filter | No | Yes | No | Opt. |
| 3.7.7.4.4 | Chip detector | No | Yes | No | Yes |
| 3.7.7.4.6 | Wear rate analysis | No | Yes | No | Opt. |

NOTE: No = Not Required for Listed Engine
Yes = Recommended Quantitative Requirement Subject to Coordination
N/R = Not Recommended
Opt = Optional (Mission-Dependent)

Figure 4.2.18-1. Comparison of Specification Requirements/455 Application.

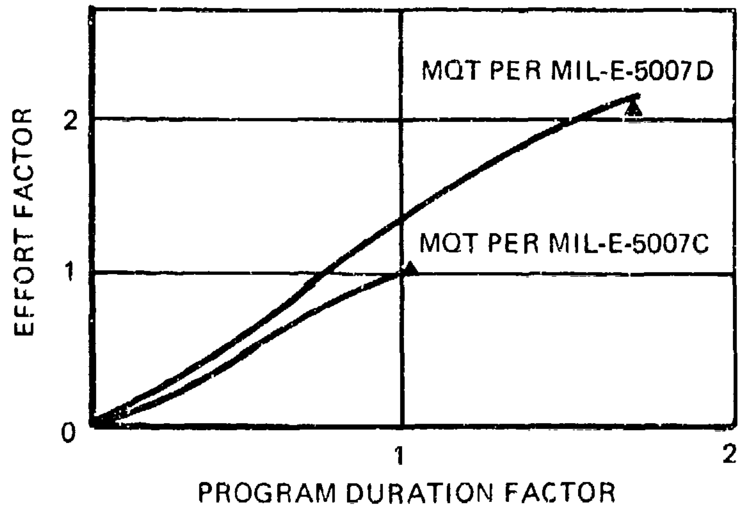


Figure 4.2.18-2. Specification Cost Impact.

system integrator determine for a specific application, that there is a need, then:

a) The quantity, composition and duration of gas products should be determined;

b) Preliminary tradeoffs should consider ordnance location versus engine inlet placement on the aircraft;

c) Mock-up tests should be conducted to measure actual values of (a); and

d) Ingestion testing should be conducted early in the engine development cycle.

A baseline engine configuration for a "worst case" gas ingestion capability could, for example, require the following additional cost items:

1) Compressor Stall/Surge Relief Valve (and Solenoid);
2) Uprated or additional primer/igniters (for relight);
3) Sensors to detect surge/flameout and provide automatic relight;
4) Fuel control module to process sensor data and command the fuel metering valve and primer/igniter.

A rough order of magnitude (ROM) estimate of added acquisition cost for the above complement is $3500/aircraft in a twin engine installation, and $350/aircraft/year in operation and support (O & S) costs. The added weight would have an airframe cost penalty (see Section 2.5). However, if airframe inlet or ordnance station changes are feasible options, they could be traded off against the foregoing costs. As a third option, an engine gas ingestion kit could be developed to equip type variations of multiple-role aircraft that have an ordnance delivery mission.

4.2.18.2 Icing; Bird Ingestion, and Sand Ingestion (3.2.5.2/3.2.5.6.1/3.2.5.6.4)

These requirements should be evaluated with the same rationale as gas ingestion because they are also mission and, to some extent, airframe/configuration dependent.

4.2.18.3 Mass Moments of Inertia; Externally Applied Forces (3.1.2.4 & 3.1.5)

These requirements can be translated into cost consequences, for a given engine and airframe, when material selection, structural analysis and weight impact are assessed.

The Mass Moments describe (but do not specify the value of) forces that an engine can impose on the airframe. The cost impact is, therefore, contained in the interactive category (see Section 2.2.5).

The Externally Applied Forces (gyroscopic moments) specify momentary and constant angular velocities which the aircraft may impose on the engine. These are specified as:

3.5 radians/second with a +/- 1g load for 15 seconds

1.4 radians/second (with various loads) for continuous duty

These requirements have initial impact on development and acquisition costs. Also, when the requirements result in added weight, they also impact airframe costs.

4.2.18.4 Bleed Air Contamination (3.1.2.11.3)

This requirement which lists permissible limits of certain contaminants (e.g., $CO_2$ @ 5000 parts/million of bleed air) should not be specified for RPV's. If safety requirements dictate its use for manned aircraft, the principal costs will accrue in development for test and demonstration purposes. However, engine bleed air may also be used for non-life support purposes (e.g., gun purge, rain removal, equipment cooling) in which case contamination levels may not be significant.

4.2.18.5 Radar Cross Section (RCS) (3.1.2.12)

This requirement can lead to unnecessary test cost if imposed on the basic (i.e., uninstalled engine). Radar reflectivity is a property of the engine inlet, which design is normally proposed by the airframe contractor and coordinated with the engine contractor. Also, RCS reduction is a tradeoff candidate. For example, an aircraft equipped with a high by-pass ratio turbofan (implies a high engine RCS) may have a lower total aircraft RCS than an aircraft with a low by-pass ratio fan and external fuel stores. The foregoing example assumes that each aircraft is designed for the same mission.

4.2.18.6 Operating Attitudes (3.2.1.5.1)

This requirement impacts the mechanical design and cost of the engine lubrication system because it sizes the scavenge capability of the oil pumps. An ROM estimate for a twin-engine UPT-type application, with a 30-second, 90 degrees (vertical) operating capability, is:

| | | |
|---|---|---|
| 1) | Scavenge Pumps/Aircraft (Additional) | 2 |
| 2) | Increase in Engine Acquisition Cost (x 2) | $700 |
| 3) | Increase in Weight (including lines and tank) | 7 pounds |
| 4) | Total A/C Acquisition Cost | $2289 |

4.2.18-7 Thrust Transients (3.2.1.5.6)

This requirement lists 18 or more precisely specified thrust transients. It should be critiqued for two reasons:

1) For some applications (e.g., RPV's), one or two thrust transients may be sufficient to describe any foreseeable requirements.

2) The "corners" of these operating transients may add unnecessary development and test cost. For example, idle to maximum thrust available is specified as seven seconds from S/L to 10K feet. Eight or nine seconds may be both adequate for the mission and readily attainable for a specific engine but demonstrating that last one or two seconds may double the cost of development and production acceptance testing.

### 4.2.19 Cost Reduction Analysis Summary

Shown in Figure 4.2.19-1 is a summary of the 18 specific cost reduction topics and their effect on the following factors:

- Change in components acquisition cost
- Change in engine weight
- Change in engine SFC
- Change in system cost due to weight for the UPT
- Change in system cost due to weight for the MMRPV
- Change in system cost due to SFC for the UPT
- Change in system cost due to SFC for the MMRPV
- System flyaway cost reduction for the UPT
- System flyaway cost reduction for the MMRPV

Note that all dollar figures shown are in 1975 dollars.

| Item | Description | Change in Component Acquisition Cost ($) | Change in Engine Weight (lbs) | Change in Engine SFC (percent) | UPT Change in System Cost due to Weight ($) | MMRPV Change in System Cost due to Weight ($) | UPT Change in System Cost due to SFC ($) | MMRPV Change in System Cost due to SFC ($) | System Flyaway Cost Reduction UPT ($) | System Flyaway Cost Reduction MMRPV ($) |
|---|---|---|---|---|---|---|---|---|---|---|
| 1 | Fan Blade Design | Figure 4.2.1-1 | Figure 4.2.1-2 | - | - | - | - | - | Figure 4.2.1-4, 5 | Figure 4.2.1-4, 5 |
| 2 | Fan Blade Materials | Figure 4.2.2-1, 3 | Page 95 | - | Page 95 | Page 96 | - | - | - | - |
| 3 | Cast Titanium Components | 393[illegible] | 7.0 UPT/ 4.2 MMRPV | - | 795 | 317 | - | - | 24,704 | 13,485 |
| 4 | Be-Ti Fan Power Shaft | Figure 4.2.4-7 | 15.84 | - | 3,595 | 1,188 | - | - | 5,802 | 2.291 |
| 5 | Brazed Axial Compressor | 351 | 8.48 | - | 1,925 | 636 | - | - | 9,350 | 4,349 |
| 6 | Hybrid Radial Compres. Diffu. | 99 | - | 1.0 | - | - | 3,313 | 880 | 4,270 | 1,557 |
| 7 | Infusion Cooled Vaporizer Plate Combustor | 2,747 | 5.40 | - | 1,226 | 405 | - | - | 8,778 | 4,181 |
| 8 | Ceramic Components | 5,619 | 15.5 | 1.87 | N/A | 1,163 | N/A | 1,646 | N/A | 8,430 |
| 9 | Powdered Metel Components | 1,554 | 11.7 | - | 2,656 | 878 | - | - | 7,740 | 2,807 |
| 10 | Elimination of LP Turbine Inlet Nozzle | 216 | 0.92 | - | 209 | 69 | - | - | 333 | 131 |
| 11 | Welded LP Turbine Assembly | | | | THIS ITEM DISCONTINUED | | | | | |
| 12 | Jet-Flap Turbine Blading | | | | THIS ITEM DEFERRED | | | | | |
| 13 | Intershaft Sealing | 100 | 0.70 | - | 159 | 53 | - | - | 434 | 190 |
| 14 | Gas Foil Bearings | | | | THIS ITEM UNDER FURTHER INVESTIGATION | | | | | |
| 15 | Simplified Lube System | Figure 4.2.15.7 | 1.56 | - | 354 | 117 | - | - | 712 | 296 |
| 16 | High Speed Accessory Drive | Figure 4.2.16-2 | Figure 4.2.16-2 | - | - | - | - | - | - | - |
| 17 | Controls and Accessories | 7,528 | 26.15 | - | 5,936 | 9,961 | - | - | 11,088 | 5,040 |
| 18 | Engine Specification Revisions | - | - | - | - | - | - | - | - | - |

NOTE: N/A = Not Applicable

Figure 4.2.19-1. Cost Reduction Analysis Summary.

# Section 5.0
# Technology Assessment

# SECTION 5.0 - TECHNOLOGY ASSESSMENT

## 5.1 Methodology; Status and Objectives

During the APSI system cost studies, many cost-influencing design decisions were evaluated. Each evaluation was characterized by literature searches, calculations, and the repetitive process of interpreting reference tables to derive cost estimates.

The airframe-company studies emphasized an awareness of the engine's airframe-cost. This awareness expands the scope of calculating LCC estimates and highlights the need to perform the important task of sensitivity-testing and design/cost iteration. In conclusion, the need for a computerized design-to-life-cost (DTLC) model became quite evident.

Teledyne CAE, therefore, reviewed the usefulness of existing life cycle cost models, including the models described in References 19 and 20, and others encountered during prior Teledyne CAE programs. All the models reviewed had one or more of the following disadvantages:

> Existing models are generally electronic-system oriented and are best applied to a system of independently removable modules (e.g., circuit boards, trays or drawers) whereas engine maintenance is a sequential process.
>
> The models assume that the equipment's operating environment is steady-state as opposed to the dynamic operating and stress environment of gas turbine engines (as characterized by the UPT composite mission described in Section 3.0).
>
> The preparation and execution of programming (because of model orientation) does not lend itself to interdisciplinary coordination. For example, an engine systems cost model should use (and interact with) the engine's computerized performance presentation to identify the cost/life influence of engine match points in terms of component loads and temperatures.
>
> The data output of most existing models provides a logistics planning function while requiring considerable interpretation for use in evaluating engine cost/benefits tradeoffs. In particular, the important cost element of engine/aircraft interaction is not addressed.

In summary, an off-the-shelf model for engine use was not available, while the work of adapting existing models to conduct frequent, timely, and credible evaluations was found to be less cost effective than the development of an engine-oriented model.

While performing the APSI cost reduction studies, Teledyne CAE developed a systems assessment methodology which lends itself to a modeling approach. The methodology which implements Teledyne CAE's proposed DTLC can be manually programmed during early program phases while a more efficient and time-saving computer program is being developed by sections or routines.

The modeling approach and logic flow are illustrated in Figure 5.1-1 and described below.

1. Mission Description - A number of mission profiles or a composite mission are described so that match points can be identified in the engine performance deck.

2. Engine/Aircraft Description - The engine of interest and its candidate air vehicles are described in an "input document". The engine parameters listed include weight, performance, and component complement (Work Unit Code structure). The aircraft parameters include weight, development/acquisition cost, number of engines, and fuel capacity.

3. Inventory Function - An inventory model is prepared to calculate engine deliveries, annual engine flying hours, and mission frequency during the build-up, steady-state and phase-out years of engine life.

4. Functional Relationships - A number of cost estimating relationships (CER's) are identified and entered. These include:

   a) Engine ownership CER's, component complement reliability (for significant components) based on thermal/mechanical stresses at discrete mission conditions (match points), with appropriate Weibull distributions, and failure mode consequences (maintainability and safety).

   b) Aircraft development and acquisition cost sensitivity to engine weight and performance variations.

5. Design-to-Cost - A Design-to-Cost (DTC) analysis is performed for the engine of interest. Cost values are entered for the significant components (particularly those that will be replaced as discrete parts during maintenance). Cost values for residual component assembly, inspection and testing are separately identified, as are the acquisition cost of the complete engine. The development cost estimate is also identified and entered.

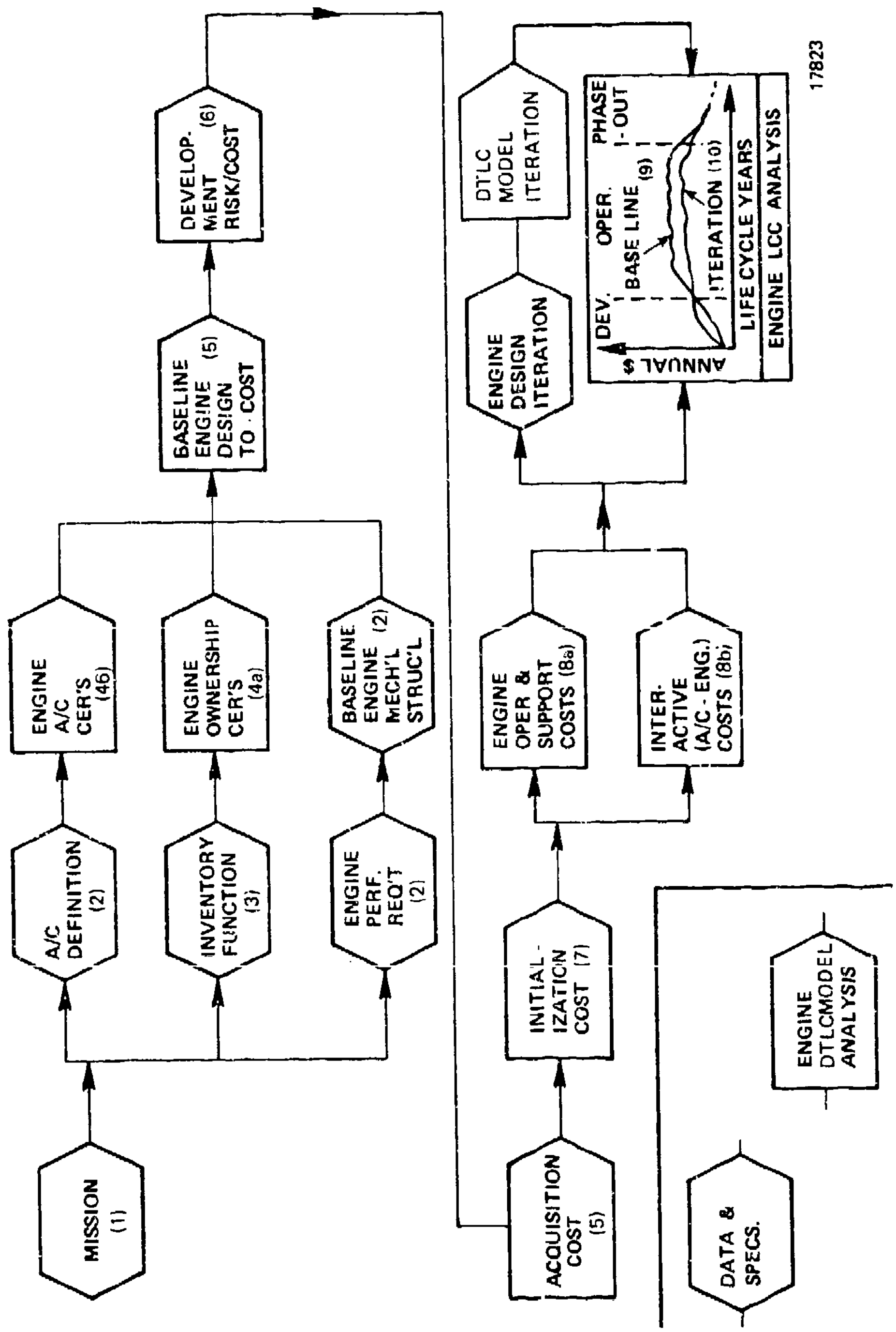


Figure 5.1-1. Engine DTLC Model - Logic Flow.

6. Development Costs - Development costs are determined for the baseline engine and proposed development program. During iterations, the development operations are treated probabilistically to address technical risk versus payoff.

7. Initialization Costs - Initialization costs are determined as a function of the proposed stocking policy and the number of parts added to DOD inventory.

8. Ownership Costs - Ownership costs are estimated in two categories as follows:

   a) Operation and Support Costs are based on reliability, maintainability and replacement parts costs (using an FMEA with extended consequences).

   b) Interactive Costs are based on the estimated cost of the baseline aircraft and the calculated effect of engine changes using the aircraft CER's.

9. Report Generation, Baseline - A systems cost estimate is prepared (manually or by executing a computer program) to identify costs of the baseline design during the program life cycle. Calculated parameters will include:

   a) Total system costs and the year in which the cost is incurred.

   b) Cost breakout to significant contributors in each year including development cost, initialization (stock set-up), parts inventory, acquisition, operations and maintenance (except for fuel) and disposal (when applicable).

   c) Interactive costs of the air vehicle/engine including fuel consumption, air vehicle development, acquisition and maintenance costs incurred as a result of the engine's weight and performance.

   d) Maintenance rates for each maintenance level.

   e) Incidence of hazardous failure modes.

10. Report Generation, Derivation and Options - Systems costs and benefits (or payoffs) are calculated for selected component improvements.

    The baseline program is executed using the parametric variations offered by the improved or derivative design. Increased costs (e.g., development) are identified, and payoffs are calculated for their expected year of accrual. Payoffs are "discounted" to identify present value in the year in which costs are initially accrued.

Where appropriate, expected ranges of parameters are derived to provide for sensitivity testing and/or probabilistic treatment.

Teledyne CAE proposed the development of this Design-to-Life Cost methodology, with computer-aided processing, in the JTDE program. Its commencement was authorized and the task is proceeding.

## 5.2 Technical Risk

In developing a method for forecasting the cost-benefit-result of design (and development) decisions, the technical risk factor and its cost impact has frequently been encountered. For example, if the technical risk is properly applied to the cost of development or to the estimate of resulting benefits, is the decision cost-effective?

Recent DOD and AFAPL RFP's have demonstrated the government's appreciation of technical risk and have promulgated methods for addressing it in design studies and tradeoff analyses. The suggested approach has been to assign a risk factor to various technical objectives of an engine development program including: SFC, thrust-to-weight; reliability growth, etc., and rank the factors as low, moderate or high. The factor assignments tend to become point estimates (explicitly or implicitly) because of the natural tendency to quantify a qualitative judgment, so that resulting estimates may be in terms of percent of risk. However, if risk is to become a decision factor in a maturing cost/benefit analysis technique, its treatment will require some mathematical rigor. That is, there will need to be established a calculus of risk.

Bierman, et al (Reference 21) have addressed the subject in detail for routine business decisions under conditions of uncertainty. Their concepts, however, appear equally valid for treating the technical risk factor in engine cost/benefit analysis.

For example, Figure 5.2-1 illustrates a decision-curve for a hypothetical engine development option, with a fixed development cost. In this example, the option is intended to achieve an LCC reducing performance improvement, and the accrued benefits (as a ratio of development cost) are treated probabilistically. In Figure 5.2-1, the resulting payoff at the expected value of 0.50 x 8-fold return = 4-fold return would tend to suggest a "go-decision" for this option.

Teledyne CAE expects to develop the definition of this subject to the point that it can be utilized in executing the DTLC model described in Section 5.1.

## 5.3 Technical Risk Estimates for Adaptive Components

In Section 5.2, the use of technical risk to evaluate cost/benefits was suggested. In that example, risk could be evaluated as a continuous

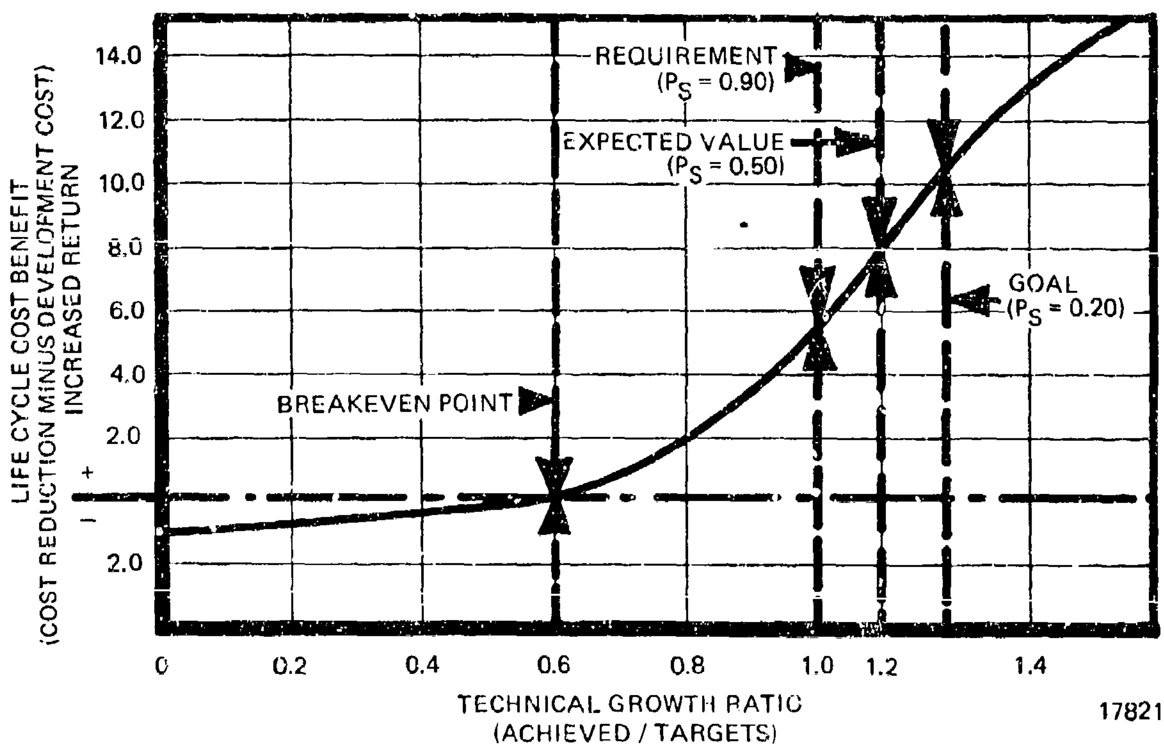


Figure 5.2-1. LCC Reduction as a Function of Development Program Success Probability.

function, increasing in proportion to technical reach. Teledyne CAE believes that approach would be appropriate where the objective can also be equated to a continuous variable (e.g., SFC, $F_n$, etc.).

A more elementry approach appears useful for single component development decisions; that is, where development success is a go/no-go proposition.

For each component, preliminary (ROM) estimates of development costs and projected benefits were developed. A single point estimate of development risk was then arrived at separately (to minimize bias) and defined as the probability that a successful component would not be derived from the development program. These two estimates were then combined to calculate a risk weighted benefit ratio (RB):

$$RB = (1 - TR)\ (SLV - CLD)/CLD \qquad (1)$$
$$BLV = (1 - TR)\ (SLV - CLD)$$

Where:

RB = Benefit Ratio
TR = Technical Risk; hence (1 - TR) = Probability of Success
BLV = Benefit (Total) value in "Datum" dollars **
CLD = Cost (Total) of development
SLV = Total Savings in System Cost (given technical success)

By applying this elementary approach to some cost reduction candidates, it was determined that some development options had more cost merit than others. If, however, the number of prospective applications increases, or if the O & S cost changes, the potential benefit ratio will also change.

In summary, it is believed that the introduction of the technical risk factor tends to make cost benefit decisions more conservative and cost effective. Also, incorporating this factor into the DTLC model (see Section 5.1) and gaining facility in its use will tend to improve the rigor and credibility of future propulsion system cost/benefit evaluations.

* "Savings" is defined as total savings in system ownership cost for all identifiable categories, adjusted to the initial period of the engine life cycle.

** "Benefit" is defined as (total savings x probability of success) - development cost; and adjusted to a "Datum" period.

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